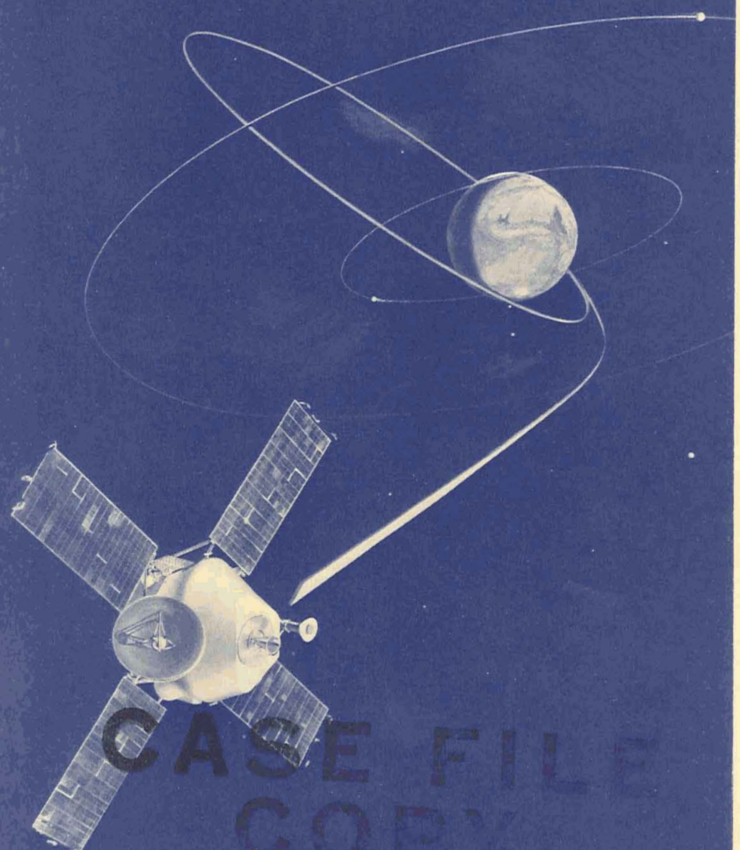
MARINER MARS 1971 PROJECT

MISSION TO MARS



JULY 1971

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

CONVERSION TABLE

To Convert	To	Multiply By
centimeters (cm)	inches (in.)	0.394
millimeters (mm)	inches (in.)	0.0394
meters (m)	inches (in.) feet (ft)	39.4 3.28
meters per second (m/sec)	feet per second (ft/sec)	3.28
kilometers (km)	feet (ft) statute miles	3280 0.621
kilometers per second (km/sec)	miles per hour (mph)	2236
kilograms (kg)	pounds (lbm)	2.20
newtons	pounds (lbf)	0.225

THE PROJECT

In May 1971, the Mariner 9 spacecraft, similar in design to that of the Mariner Mars 1969 spacecraft (Mariners 6 and 7), was launched by an Atlas/Centaur vehicle on an orbital mission to Mars to study the characteristics of that planet for a period of at least 90 days. Mars arrival time for the spacecraft is November 14, 1971. The spacecraft will remain in orbit about the planet Mars for a minimum of 17 years.

During the first 90 days that the spacecraft is in orbit, about 70 percent of the planet's surface will be mapped; the composition, density, pressure, and temperature of the atmosphere and the structure, temperature, and composition of the surface will be studied. Data on the changes in surface markings, such as the seasonal darkening observed during Martian spring, will be obtained.

Mariner 9 will provide almost twelve times more planetary data than all previous planetary missions combined and will demonstrate the ability to perform a type of orbital operation in which information from any orbital pass can be used to develop an operations plan for the rest of the orbital passes.

The success of the Mariner Mars 1971 mission depends primarily upon the Spacecraft System, for the actual spacecraft that performs precisely timed program and ground commanded activities; the Tracking and Data System, for tracking the spacecraft, transmitting the necessary commands to each spacecraft, and receiving and transmitting information between the tracking stations and the Space Flight Operations Facility; and the Mission Operations System, for commanding and controlling each spacecraft during the phases of cruise and orbit.

These systems are integrated under the overall management of the Jet Propulsion Laboratory, California Institute of Technology, as directed by the NASA Office of Space Science and Applications. NASA's Lewis Research Center was responsible for the Launch Vehicle System which placed the Mariner 9 spacecraft in the desired trajectory to Mars. The three remaining systems are under the management of the Jet Propulsion Laboratory (see Figure 1).

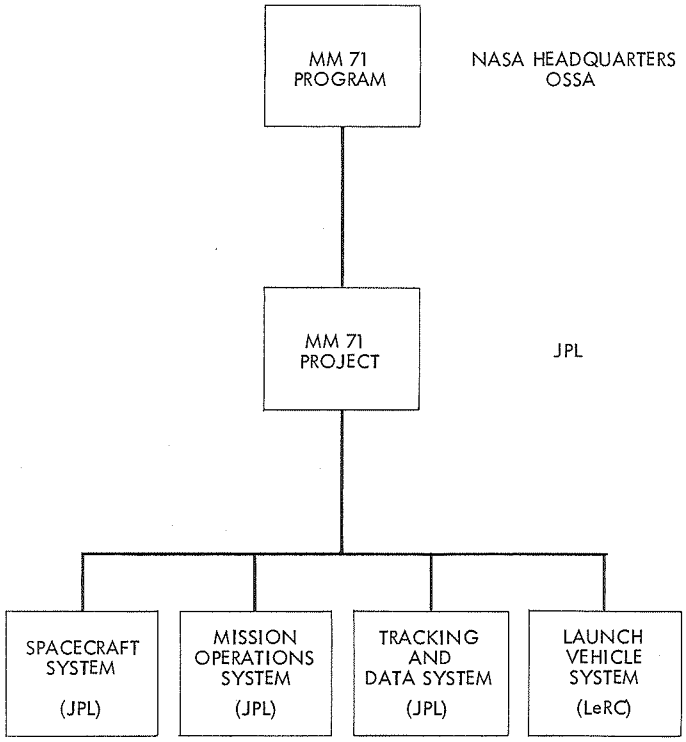


Figure 1. Project Organization

SCIENTIFIC GOALS AND EXPECTATIONS

Until Mariner 4 flew past Mars in 1965, all of man's knowledge of that planet had been obtained through Earth-based observations by using telescopes. During the past 100 years, Mars has been revealed as a planet in some ways not unlike the Earth and in other ways completely different. Data from previous flyby missions to Mars have indicated some similarity between the two planets, but more often have emphasized the differences: the very low Martian temperatures at certain latitudes and at night; the rugged terrain with linear features and craters similar in appearance to those on the Moon; the thin, predominantly carbon dioxide atmosphere; and low surface pressure. Mariners 4, 6, and 7 could make only "spotchecks" of Mars because they were in the planet's vicinity for a very short time. To accumulate data for longer time periods and to observe changes near and on the surface, the spacecraft must orbit the planet.

Six experiments, which are described in the subsequent paragraphs, were selected for the Mariner 9 orbiter. The instrument and Principal Investigator (or Principal Investigators) associated with these experiments are given in Table 1.

Table 1. Mariner 9 Science Experiments

Experiment	Instrument	Principal Investigator
Television	Television cameras	Mr. H. Masursky,[b] United States Geological Survey, Flagstaff Dr. G. Briggs, Bellcomm Inc.* Dr. G. de Vaucouleurs, University of Texas Dr. J. Lederberg, Stanford University Mr. B. Smith, New Mexico State University
Ultraviolet spectroscopy	Ultraviolet spectrometer	Dr. C. Barth, University of Colorado
Infrared spectroscopy	Infrared interferometer spectrometer	Dr. R. Hanel, Goddard Space Flight Center
Infrared radiometry	Infrared radiometer	Dr. G. Neugebauer, California Institute of Technology
S-band occultation	None[a]	Dr. A. Kliore, JPL
Celestial mechanics	None[a]	Mr. J. Lorell,[b] JPL Dr. I. Shapiro, Massachusetts Institute of Technology

[a] This experiment uses the spacecraft's radio subsystem to obtain data. No additional instrument is used.

[b] Team Leader.

*Dr. Briggs, JPL employee effective September 1971.

Television Experiment

The primary objective of the television experiment is to provide data to increase the scientific knowledge regarding Mars and the solar system or, as defined in more detail:

(1) To investigate various Martian phenomena to achieve a better understanding of the dynamics, history, environment, and surface physiography of Mars.

(2) To obtain pictures suitable for development of better geologic, dynamic, and topographic maps of the planet.

The experiment probably will provide no direct evidence of the possibility of life on Mars, but it is expected that it will provide indirect evidence regarding the suitability of the planet as a possible environment for life.

Two types of scientific investigations will be conducted: fixed features and variable features. The fixed-features investigation will allow the mapping of surface features at greater resolutions (finer detail) than are possible from Earth-based (telescope) observations or those made from previous spacecraft. The objectives are:

(1) To obtain a broad range of information for investigating structural features, crater configuration and distribution, and local surface conditions.

(2) To determine surface slopes and elevations and surface brightness and albedo (reflective power) differences, and to improve the accuracy of the photometric function.

(3) To obtain improved values for the shape of the planet.

(4) To study the surface characteristics of the Martian satellites, Phobos and Deimos.

The variable-features investigation will provide information on the time-variable surface markings and on atmospheric structure and circulation, diurnal and seasonal changes, and the possibility of life on Mars. Specific phenomena to be studied are:

(1) Seasonal darkening variations of the surface.
(2) Polar-cap and cap-edge phenomena.
(3) Nightside atmospheric and surface fluorescence.
(4) Atmospheric haze.
(5) White "clouds" in nonpolar regions.
(6) Dust clouds and dust storms.

The television subsystem used in this experiment consists of two television cameras (wide and narrow angle) mounted on the spacecraft's planetary scan platform. The camera optics and some parts of the supporting electronics are identical to the equipment used on Mariners 6 and 7. The wide-angle camera (see Figure 2) has a rectangular field of view of 11 by 14 degrees and a focal length of 50 millimeters. The narrow-angle camera (see Figure 3) has a rectangular field of view of 1.1 by 1.4 degrees and a focal length of 500 millimeters. The resolution of surface features and of the area in the field of view of each camera is dependent on the line-of-sight range from the cameras to the planet's surface. With the cameras pointed in the nadir direction (looking vertically downward at the surface) and the spacecraft at an altitude of 1250 km, the wide- and narrow-angle cameras can detect objects or features under about 1 km and 0.1 km, respectively. Television camera performance is summarized in Table 2.

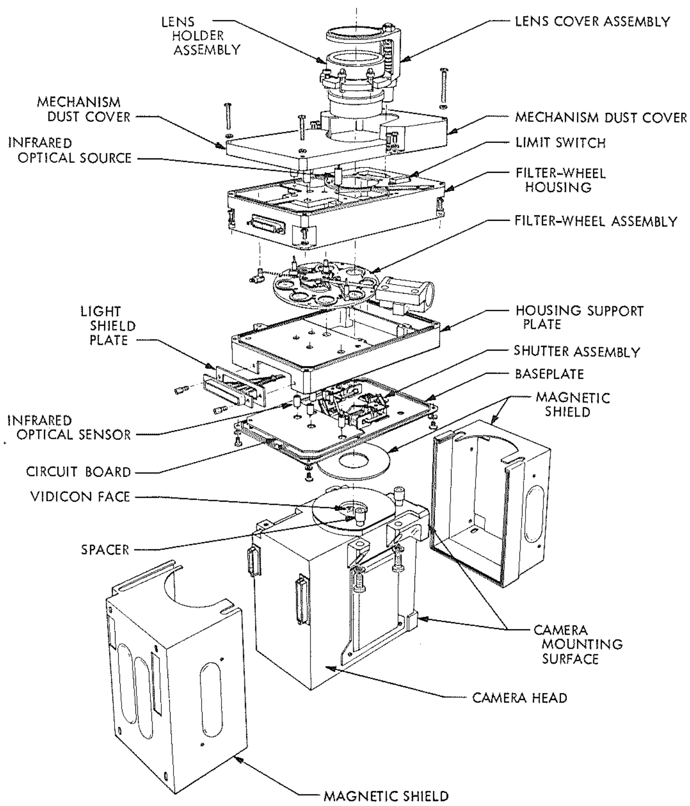


Figure 2. Wide-Angle Television Camera

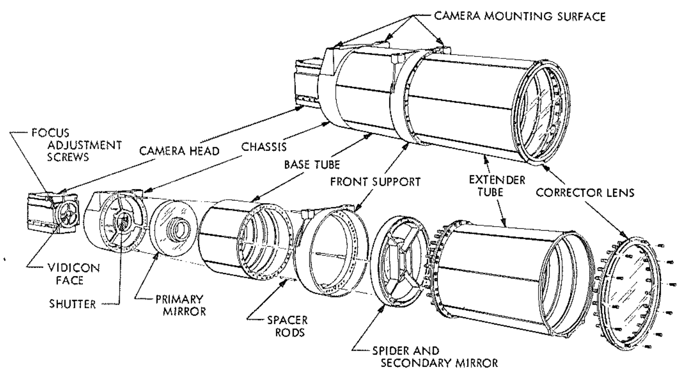


Figure 3. Narrow-Angle Television Camera

Table 2. Camera Performance

Characteristic	Wide-Angle Camera	Narrow-Angle Camera
Focal length, millimeters	50	500
Focal ratio	f/4.0	f/2.35
Shutter speed range, milliseconds	3 to 6144	3 to 6144
Automatic shutter speeds, milliseconds	48, 96, 192	6, 12, 24
Angular field of view, degrees	11 x 14	1.1 x 1.4
Active vidicon target raster, millimeters	9.6 x 12.5	9.6 x 12.5
Scan lines per frame	700	700
Picture elements per line	832	832
Bits/picture element	9	9
Frame time, seconds	42	42

Ultraviolet Spectroscopy Experiment

The ultraviolet spectroscopy experiment will allow the surface and lower atmosphere to be observed in the ultraviolet spectral region. The experiment measurements will help to determine:

(1) Local atmospheric pressure over most of the planet, using the scattering properties of the lower atmosphere.

(2) Local ozone concentration.

(3) Photometric function in the near ultraviolet.

Data will provide clues to the seasonal darkening, white and yellow clouds, and the blue haze/blue clearing. Areas of high ozone concentration may indicate where to look for suitable micro-environments for life.

The experiment also will provide data on:

(1) Composition and structure of the upper atmosphere as a function of latitude, longitude, and time.

(2) Ionospheric composition and its variations.

(3) Distribution and escape rate of atomic hydrogen from the exosphere.

(4) Distribution and variability of ultraviolet aurora to explore potential induced magnetic fields.

Intensities in the near-planet region will be monitored to the limits of the exosphere (outermost layer of the planet's atmosphere).

The instrument used in this experiment is an Ebert-Fasti type of spectrometer, through which ultraviolet light enters and is split into its component wavelengths by a reflection diffraction grating. Two exit slits allow two measurement channels. The detectors are photomultiplier tubes with photocathode and window materials that provide additional wide-range wavelength discrimination.

The ultraviolet spectrometer on Mariner 9 (see Figure 4) is basically the same as that on Mariners 6 and 7, with some modifications. The channel 1 photomultiplier tube (F) has a spectral range of 1450 to 3500 angstroms. (One angstrom unit is equal to 10^{-8} cm, about the size of an atomic nucleus.) A step gain amplifier incorporated with this channel provides control over the expected range of surface brightness. The resolution has been maximized by reducing the field of view from 0.25 by 2.5 degrees to 0.20 by 0.55 degree. The channel 2 photomultiplier tube (G) has a spectral range from

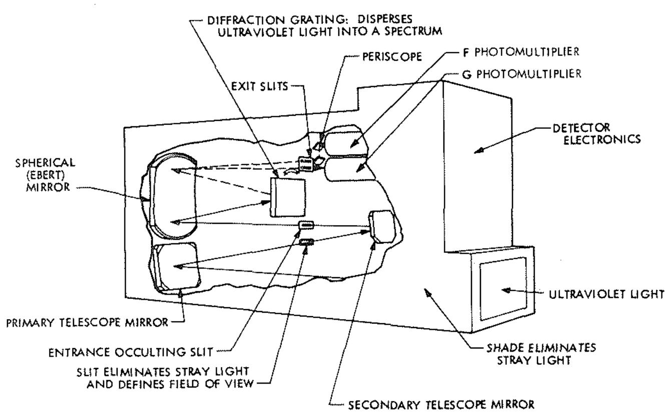


Figure 4. Ultraviolet Spectrometer

1100 to 1900 angstroms and a field of view of 0.20 by 2.0 degrees. One spectral sweep will be recorded each 3 seconds.

Infrared Spectroscopy Experiment

The infrared spectroscopy experiment uses measurements of the spectral radiance of the radiation emitted from the Martian atmosphere and surface to determine atmospheric and surface parameters. These parameters will be used in investigations of the physical behavior of the atmosphere and of the composition and structure of the surface.

The experiment will provide measurements of the vertical temperature structure, composition, and dynamics of the atmosphere. The total content of water in the atmosphere, variations of water vapor, and measurements of the minor atmospheric constituents will be derived from the data obtained. Values for the temperature, composition, and thermal properties of the surface, including the polar caps, also will be derived from the spectra.

The instrument used in this experiment (see Figure 5) is a Michelson interferometer spectrometer similar to that used for the Nimbus III and IV meteorological Earth satellites, with modifications made in the mechanical and electrical components. An essential part of the instrument is the beamsplitter, which divides the incoming radiation into two approximately equal components. After reflection from the fixed and moving mirrors, respectively, the two beams "interfere with each other" with a phase difference between both beams. The recombined components are focused on the detector, where the intensity is recorded as a function of path difference.

The spectral range covered by the Mariner 9 instrument is 200 to 1600 cm^{-1} (50 to 6 microns), with 2.4-cm^{-1}-wide

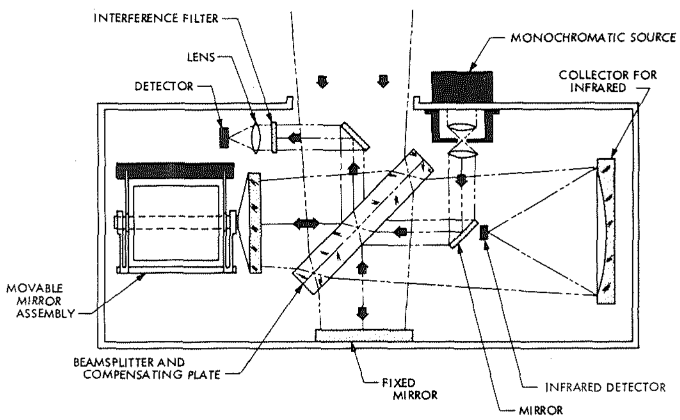


Figure 5. Infrared Interferometer Spectrometer

spectral resolution elements. The spatial resolution is about 100 km for an altitude of 1250 km with the 4.5-degree conical field of view. One spectrum (interferogram) will be recorded each 21 seconds.

Infrared Radiometry Experiment

The scientific objective of the infrared radiometry experiment is to measure the temperatures of the soil over a wide coverage of the planet's surface, including the dark side of the planet from terminator (dividing line between lighted and unlighted parts of planet's surface) to limb. Information will be obtained on:

(1) Thermophysical properties of the surface materials.
(2) Occurrence of irregularities in the cooling curve, other than those expected from albedo variations.
(3) Existence of "hot spots," which may indicate sources of internal heat.

The Mariner 9 infrared radiometer (see Figure 6) will provide brightness temperatures of the Martian surface by measuring the energy radiated in the 8- to 12-micron and 18- to 25-micron wavelength bands. (One micron is equal to 10,000 angstroms.) By using refractive optics, infrared radiation is focused on detectors, which use 13-junction bismuth-antimony thermopiles, in two independent channels. The channels have fields of view of 0.6 and 0.7 degree, respectively, and provide resolutions of about 13 and 15 km at a range of 1250 km. Although the Mariner 9 radiometer is basically the same as that flown on Mariners 6 and 7, it has been modified to provide clearer definition of the fields of view.

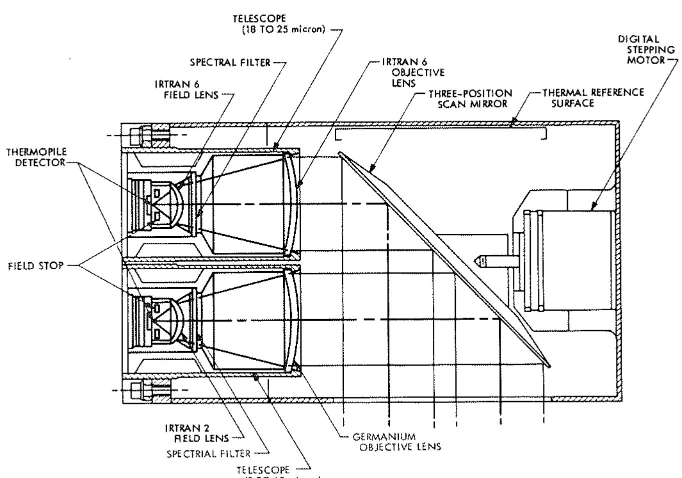


Figure 6. Infrared Radiometer

S-Band Occultation Experiment

The S-band occultation experiment uses the distortion of radio waves as they pass through the Martian atmosphere. The occultations occur when the spacecraft's orbital path carries the spacecraft in back of the planet as observed from Earth. Measurements can be made twice during each orbital revolution, once when the spacecraft disappears behind Mars and once when it reappears.

Information will be obtained on the shape of the planet and on variations in the properties of the atmosphere with latitude, season, and time of Martian day. Measurements of atmospheric density and pressure at many points above the surface will substantiate previous measurements, define possible variations, and provide data for the establishment of a circulation model of the atmosphere. Measurements of the ionosphere, made with varying Sun angles, will lead to a better understanding of the photochemical processes and reactions in the upper atmosphere.

No specific instrument is used to obtain these measurements. The S-band occultation experiment uses only the spacecraft's radio subsystem.

Celestial Mechanics Experiment

Like the S-band occultation experiment, no specific instrument is used to make measurements. Data are obtained by using the spacecraft's radio subsystem. Through observations of the motions of the spacecraft, celestial mechanics information is obtained to help to determine the size, shape, distance, and position of Mars and to detect large concentrations of mass (mascons) on the planet. By long-term observations, the experiment will provide a refined test of Einstein's theory of relativity and will study variations in the Martian gravity field.

MISSION DESCRIPTION

Selecting the best trajectory (flight path) between Earth and Mars is similar to selecting the best way to build a railroad when the choice lies between shorter but steeper routes requiring larger locomotives, and longer but more level routes that may take longer to travel. For the Mariner Mars 1971 orbiter, the choice of trajectories between Earth and Mars is governed by various factors relating to the launch phase, the interplanetary transit phase, and the Mars approach phase. The exact flight path chosen for any planetary flight depends upon the day of the month and the time of day of launch. The velocity required to reach Mars is lowest when Earth launch and Mars arrival occur almost on opposite sides of the Sun. Such conditions prevail only for a few weeks each 25 months.

For a spacecraft to follow a trajectory that will result in an encounter with Mars, it must escape from the gravitational field of the Earth and become a satellite of the Sun. The direction and velocity at which the spacecraft must enter this elliptical path (heliocentric) about the Sun is determined by the relative position of the Earth at the time of launch and Mars at the time of arrival. An Earth-Sun-Mars situation occurs in 1971 that offers a favorable opportunity for design of Earth-to-Mars trajectories because the velocity required to inject the spacecraft is lower than at any time during the next 15-year cycle of opportunities. The next equally favorable opportunity will not occur until 1986.

The direction in which the spacecraft must leave the Earth in order to follow the required trajectory to Mars also is unusually favorable. Mariner Mars 1971 utilized a direct-ascent mode in which the motor of each stage was started immediately after separation from the previous stage. This mode results in greater spacecraft weight or scientific payload capability, greater reliability, and more favorable tracking coverage.

The 26-m (85-ft)-diameter antennas located at tracking stations around the world are providing 24-hour coverage of the flight, and serve as backup to the single 64-m (210-ft) antenna at Goldstone, California, for data transmission to Earth. The entire 26-m antenna network (three stations) can return about half of the tape-recorded data that the single 64-m antenna can return.

Mariner 9 will map 70% of the Martian surface within 90 days and will cover some areas again on a 17-day cycle to see if any dynamic changes have occurred. The TV system will map the planet with contiguous low-resolution (1 km per TV line) photographs and with high-resolution (100 m per TV line) spot coverage dispersed to obtain information on targets of greatest interest.

Launch Phase (Liftoff to Injection Into Earth-Mars Trajectory)

The Mariner 9 spacecraft was launched from Cape Kennedy, Florida, at 22 hours 23 minutes GMT (3:23 P.M. PDT) May 30,1971. The launch azimuth (the angle measured clockwise from due north to the intended direction of launch) was 92.7 degrees.

After injection, the spacecraft separated from the launch vehicle, extended its solar panels, and automatically acquired the sun and the star, Canopus, to provide external references for three-axis stabilization.

Mariner 9 has been tracked continuously since injection by the Tracking and Data System to obtain the telemetry and tracking data necessary to determine the spacecraft trajectory.

Maneuver Phase (When Interplanetary Trajectory-Correction Maneuvers are Executed by Spacecraft)

The first trajectory-correction maneuver was executed five days after launch at 00 hours 22 minutes GMT on June 5, 1971 (5:22P.M.PDT on June 4,1971).The maneuver changed spacecraft velocity by 6.7 m/sec to simultaneously remove the intentional bias in the launch aim point and to correct the launch vehicle injection error. The maneuver aim point was selected to achieve an inclination to the Mars equator of 65 degrees and a miss distance of 8200 km from the center of Mars, to provide an optimal transfer from the approach trajectory to the desired Mars orbit upon arrival. The accuracy of this first maneuver will not be fully determined until a few weeks before arrival at Mars. At that time, a second correction maneuver will be executed, if required.

Cruise Phase (Injection to Start of Pre-Insertion Phase, Excluding Trajectory Correction)

During cruise, the spacecraft will be tracked alternately from deep space stations around the world. During the cruise phase, engineering telemetry data, which will indicate the condition of the spacecraft, will be transmitted continuously to Earth. Figure 7 shows the heliocentric view of the spacecraft during cruise.

Pre-Insertion Phase (About 5 Days to 1 Day Before Insertion into Mars Orbit)

A few days before insertion, the spacecraft will be close enough to the planet to take scientific measurements, primarily television pictures. Data will be recorded on-board the spacecraft in the data storage subsystem tape recorder and subsequently played back to the 64-m-diameter antenna at the Goldstone, California, tracking station.

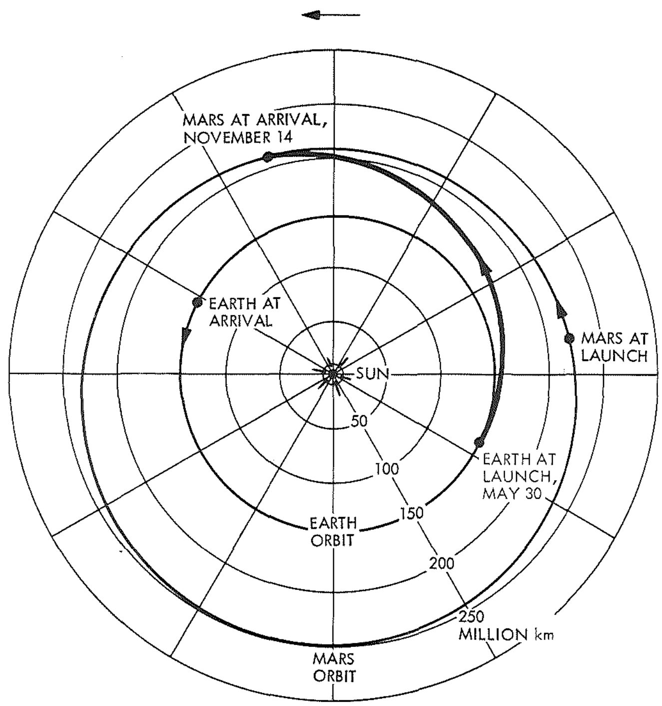


Figure 7. Heliocentric View of Spacecraft

Orbital Insertion and Trim Phase (1 Day Before Arrival Through Orbital Insertion to End of Trim Maneuvers)

The spacecraft will arrive at Mars about 00 hours 05 minutes GMT on November 14, 1971 (4:05 P.M. PST, November 13, 1971). The arrival time, which is controlled by the interplanetary trajectory-correction maneuvers, was selected to occur when Mars is in view of the 64-m Goldstone antenna to place the orbit insertion over Goldstone. This arrival time also allows a single trim maneuver to be executed over Goldstone 2 to 8 days later to achieve the desired time phasing of the final Mars orbit. Executing maneuvers over Goldstone enables engineering telemetry (spacecraft information) to be received via the 64-m antenna. The Mariner 9 rocket motor will be started 15 minutes before arrival and will be fired for about 15 minutes until the spacecraft has been slowed down by approximately 1600 m/sec (see Figure 8). After engine shutdown, the spacecraft will be in an elliptical orbit around Mars with a 12.5-hour nominal period, a nominal periapsis (point closest

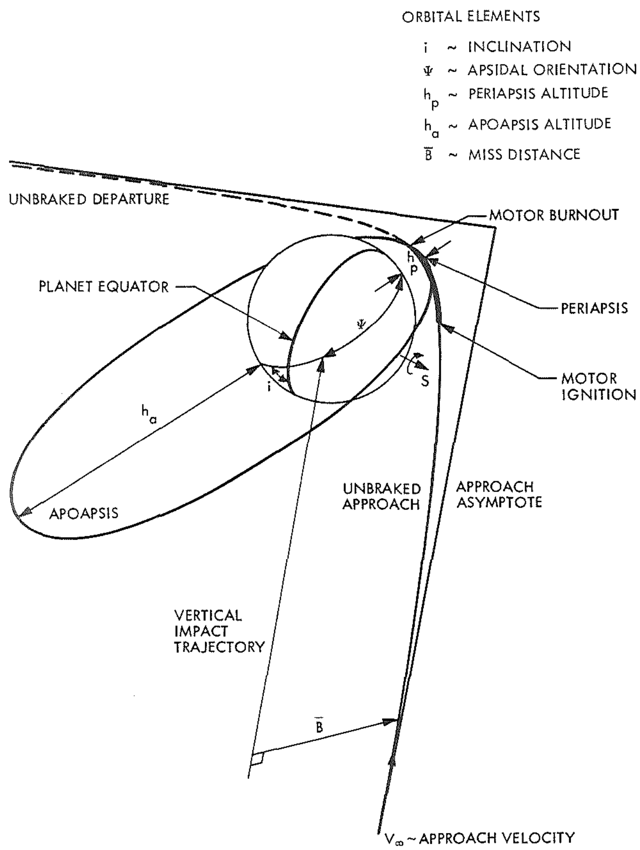


Figure 8. Approach Trajectory and Elliptical Orbit

to Mars) altitude of 1300 km, and a nominal apoapsis (point farthest from Mars) altitude of 17,917 km. The nominal 1300-km periapsis altitude ensures that the minimum altitude achieved by the orbital insertion maneuver will not be below 1200 km, and it will not be necessary to perform an orbit trim maneuver to correct the periapsis altitude. When periapsis passage coincides with Goldstone zenith, a single orbit trim maneuver will be executed to change the orbital period to the final desired value of 11.98 hours.

Orbital Phase

The trim maneuver will simultaneously adjust the orbital period and time phasing to enable Mariner 9 to playback its fully loaded tape recorder after each Goldstone rise, record another load of data near periapsis, and playback this second load before Goldstone set. The recorder will then be ready to record data at the next periapsis passage 11.98 hours later, when Mariner 9 is not visible at Goldstone. In the event of large insertion dispersions, two orbit trim maneuvers will be executed to achieve synchronism. The first will be performed at or near periapsis two days after insertion and the second, at the periapsis eight days after insertion.

Orbit Characteristics

The orbit characteristics of Mariner 9 are summarized in Table 3.

Table 3. Proposed Orbit Characteristics

Period	11.98 hr
Periapsis altitude	1200 to 1500 km
Inclination	65 deg
Apsidal orientation	136 to 142 deg
Arrival date	November 14, 1971(GMT)

The 11.98-hour orbital period allows variable feature studies of Mars to be made. Mars revolves about the Sun at the rate of 0.538 degrees of celestial longitude per Mars mean solar day (24.660 hours). The 11.98 hour-period is 17/35 of the Martian mean solar day. Thus, after 17 Martian mean solar days (and 35 spacecraft revolutions) the orbit ground track on the surface of Mars will begin to repeat itself. The 11.98-hour orbit period is synchronized with the 0.538-degree/day motion of Mars about the Sun. After every 17 Martian mean solar days, the solar illumination conditions of any specific point on the planet as viewed from orbit will be nearly constant. This is important for studies of variable features.

Because the rotational period of Mars is slightly greater than 24 hours, the planet will rotate slightly less than 180 degrees during each spacecraft orbital revolution of 11.98 hours. Thus, science coverage will occur on alternate sides of the planet with a small 9- to 10-degree longitude shift per spacecraft pass. This initial longitude circuit will cover latitudes from about 65 degrees to about 20 degrees south (see Figure 9). After the initial 17 to 18 day circuit has been made for the mapping pictures, the motion of the planet about the Sun will move the evening terminator so that the picture track can be moved north.

The minimum periapsis altitude of 1200 km or higher was chosen to ensure that, when vertical wide-angle camera another load of data near periapsis, and playback this second load before Goldstone set. The recorder will then be ready to record data at the next periapsis passage 11.98 hours later, when Mariner 9 is not visible at Goldstone. In the event of large insertion dispersions, two orbit trim maneuvers will be executed to achieve synchronism. The first will be performed at or near periapsis two days after insertion and the second, at the periapsis eight days after insertion.

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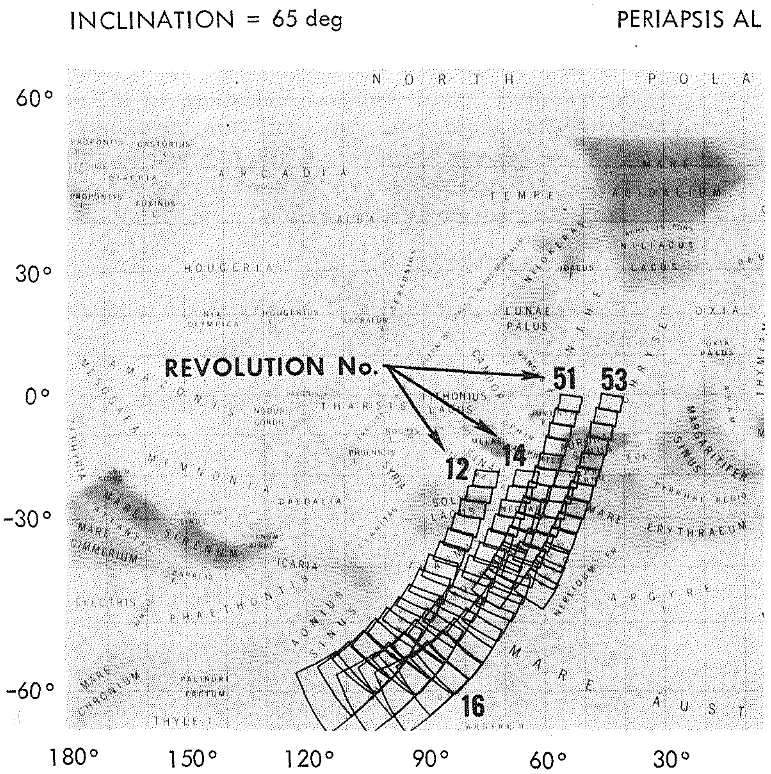


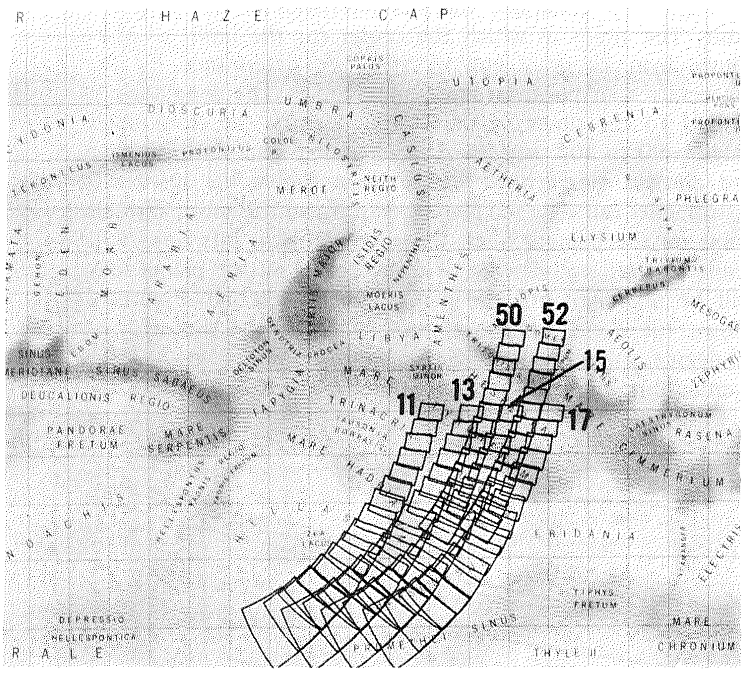
Figure 9. Typical Wide-Angle Camera Footprints for Surface Mapping

pictures are taken at periapsis, there will be front-lap (contiguity) between adjacent pictures taken every 84 sec; side-lap gaps will occur between sequences of pictures taken near periapsis on adjacent orbits. However, these gaps will be filled in on subsequent 17- to 18-day longitude circuits.

The spacecraft inclination is 65 degrees. This is a compromise between a higher inclination, which would provide excellent coverage of the South polar region, and a lower inclination, which would provide better viewing of variable-features phenomena near the sub-solar point. Planetary quarantine does not affect the choice of inclination.

After November 14, the Mars-to-Earth communication distance continues to increase slowly. For a given transmitted data rate, the error rate in the information radioed to Earth (bit error) will also increase slowly. Unless the bit error rate rises above 5 x 10^{-3}, the maximum mission data rate of 16.2 kilobits/sec will be maintained. Once this bit error rate is exceeded, it may be necessary to reduce the maximum data rate to 8.1 kilobits/sec. Because Goldstone is the only receiving station equipped to handle the 16.2 or 8.1 kilobit/sec

TITUDE = 1,250 Km PERIOD = 11.9765 hr



0° 330° 300° 270° 240° 210° 180°

data rates, the earliest arrival on November 14 will maximize data return. A later arrival date would reduce the total number of days during which the 16.2 kilobit/sec data rate could be maintained.

The apsidal orientation angle, ψ, is measured from the incoming hyperbolic direction in which the spacecraft approaches Mars (the approach asymptote) to the orbit ellipse line of apsides (see Figure 8). As ψ is increased, the latitude of periapsis moves north on the surface of Mars, and the opportunities for global TV coverage are enhanced. Global TV coverage consists of several low resolution pictures which map the lighted disk of the planet about 2 hours before periapsis. The time is dictated by the scan platform viewing limitations. If ψ is increased, pictures can be taken at higher altitudes, which reduces the total number of pictures needed for a mosaic of the lighted disk.

If a periapsis-to-periapsis (minimum impulsive) orbit insertion is attempted, the resulting apsidal orientation angle will be equal to about 118 degrees. However, by consuming more spacecraft retro fuel at orbit insertion, the value of ψ can be increased to a maximum of 136 to 142 degrees, depending on the delivery accuracy of the spacecraft at the planet.

Figure 10 illustrates the characteristics of the proposed orbit. The times at which the center (or the limb) of Mars comes into or goes out of view are determined by the spacecraft scan platform mechanical limits. The cone angle is limited to the range of 96 to 165 degrees; the clock angle range is 90 to 305 degrees. As long as the scan platform cone and clock angles remain within these limits, the spacecraft instruments can view the planet vertically at the sub-spacecraft point on the planet. When the scan platform hits one of its limits, off-vertical viewing of some points on the planet may still be possible until the planet limb is reached. Figure 10 shows the time when the spacecraft passes through the 60-degree solar incidence angle contour on its way to the evening terminator (90-degree incidence angle). Because of the

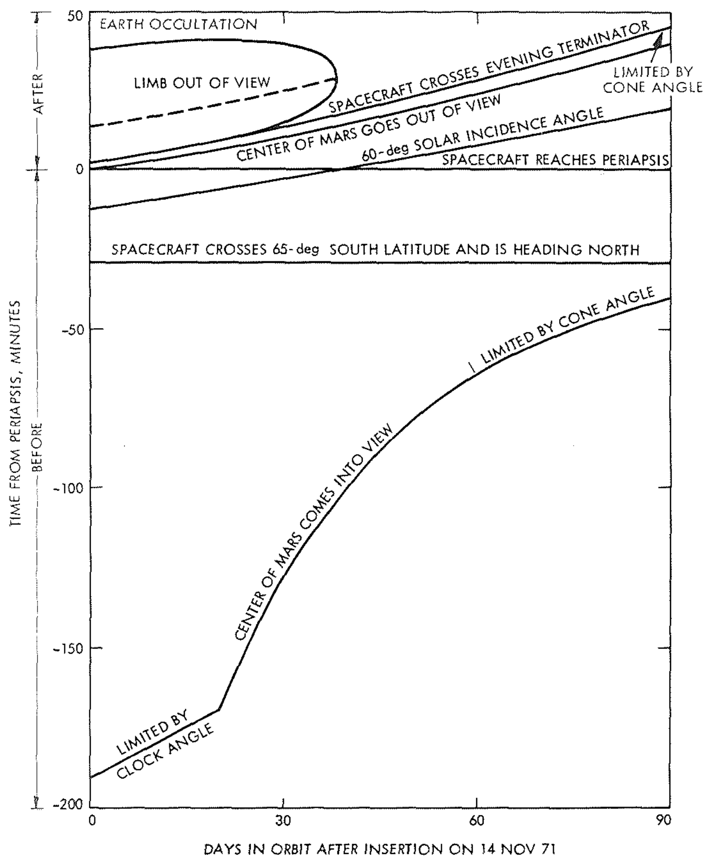


Figure 10. Orbit Characteristics

orbit inclination selected, the spacecraft ground track will be confined between 65 degrees north and 65 degrees south latitude on Mars. About 29 minutes before periapsis, the spacecraft crosses 65 degrees south latitude and starts to head north along its ground track towards periapsis.

Shortly after passing through orbit periapsis, the spacecraft enters Earth occultation, as shown in Figure 10. During occultation, the planet limb will pass out of view of the scan platform as shown by the dashed line in the figure. After approximately 38 days from insertion, Earth occultations will cease for the balance of the 90-day nominal mission.

There will be no solar occultations during the 90-day nominal mission. Solar occultations will commence about 135 days after orbit insertion (during the proposed extended mission). The occultation period will not exceed 1 hour and 40 minutes on any orbit. The solar occultations will end about 190 days after insertion.

An additional period of Earth occultations will commence about 170 days after insertion and last for about 40 days. Reflected stray sunlight from the surface of Mars may enter the field of view of the Canopus sensor for several minutes during each orbit starting some 40 days after orbit insertion. The spacecraft must turn on its gyros to maintain its roll reference during these periods. Electrical power drain will be increased.

Science Sequences

Figures 11 and 12 show typical Goldstone zenith and nadir science sequences as they might appear early in the mission. As Mars rises to an elevation angle of approximately 15 degrees above the local horizon at Goldstone, playback of the tape-load of data taken during the preceding nadir pass will begin. Nearly 3 hours are required to play back the data at a rate of 16.2 kilobits/sec. After playback, there may be an opportunity to take global coverage pictures.

Three geodesy pictures from the wide-angle camera will be taken at 84-sec intervals beginning about 1 hour and 30 minutes before periapsis. These wide-angle camera pictures will be targeted at latitudes of about 15 degrees south, 30 degrees south, and 45 degrees south. On one pair of orbits, the TV pictures will be taken at the sub-spacecraft longitude; on the succeeding pair of orbits, the pictures will be taken ahead (east) of the sub-spacecraft longitude to provide a stereo effect. The three geodesy pictures will then be played back. Figures 13 and 14 show typical geodesy pictures.

A scan platform cone and clock angle will be selected so that, as the ultraviolet spectrometer slit sweeps down through the atmosphere above the brightest portion of the Mars

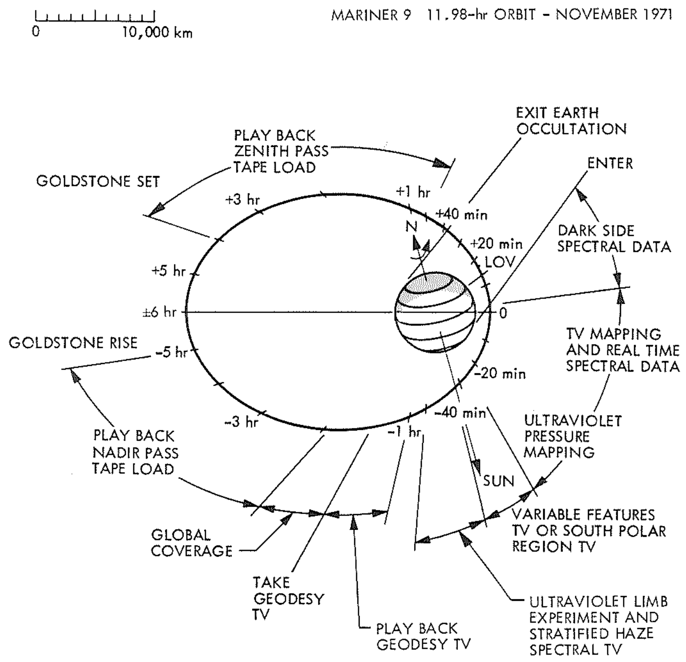


Figure 11. Typical Goldstone Zenith Science Sequence

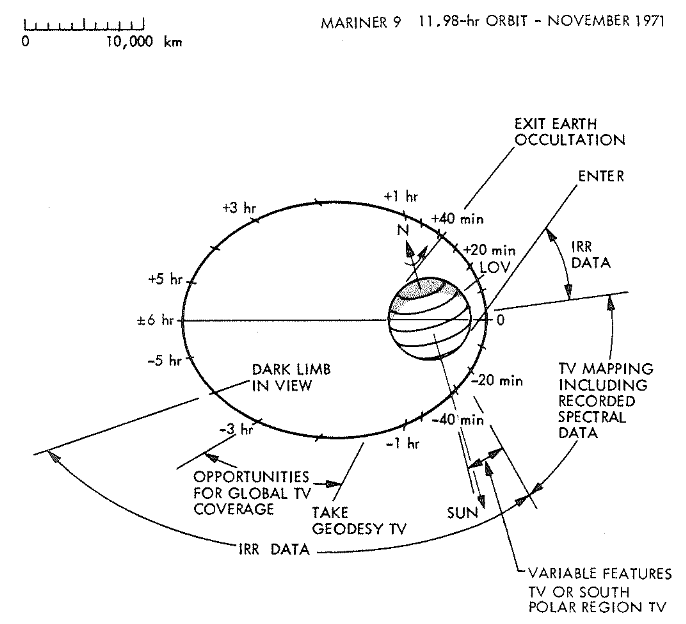


Figure 12. Typical Goldstone Nadir Science Sequence

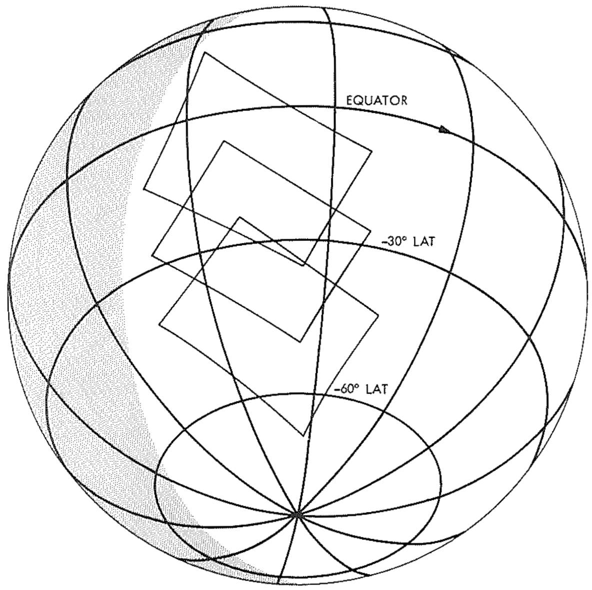


Figure 13. Typical Geodesy Sequence, View Near Vertical

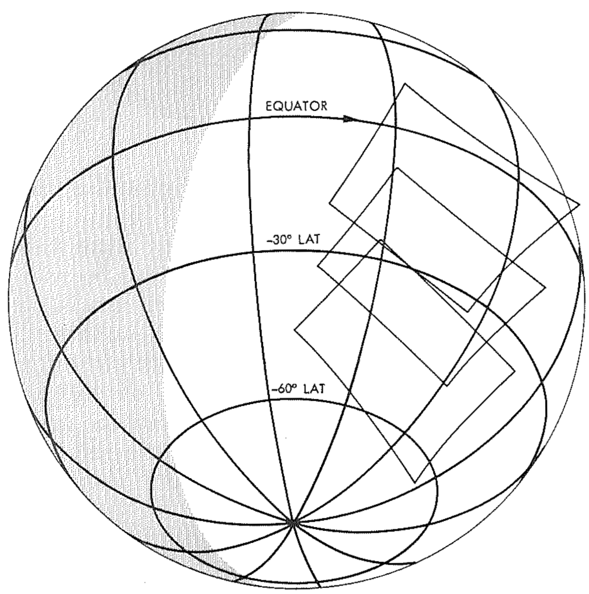


Figure 14. Typical Geodesy Sequence, Looking Eastward

surface, the slit will be as close to perpendicular to the local vertical as is possible, when it passes through an altitude of 100 km above the surface (see Figure 15). The ultraviolet spectrometer data is transmitted in real time at 8.1 kilobits/sec. During the conduct of this bright limb experiment, three

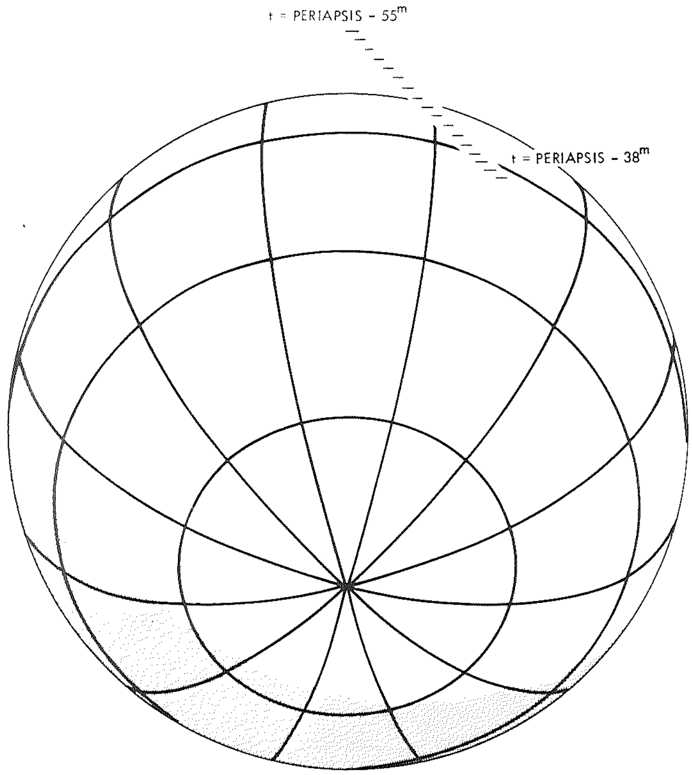


Figure 15. View of the Ultraviolet Spectroscopy Experiment

TV pictures of the atmosphere immediately above the planet limb may be recorded. On succeeding orbits, wide-angle camera pictures using different colored filters may be taken to provide spectral studies of stratified atmospheric haze.

Approximately four wide-or narrow-angle pictures may be taken when the spacecraft is orbiting near 65 degrees south latitude. These pictures may be aimed toward the south polar region or, to study variable features on the surface, north (on alternate orbits) toward the latitudes where the solar illumination is highest.

Ultraviolet pressure mapping will be conducted by pointing the ultraviolet spectrometer at the lighted side of the planet surface during a portion of the orbit up to within about 20 degrees of the terminator. The data will be transmitted in real time at 8.1 kilobits/sec.

The mapping sequence is the heart of the mission. This sequence consists largely of TV wide-angle camera pictures of selected points on the Mars surface supplemented by a few TV narrow-angle camera pictures at ten times better resolution.

Figures 9 and 16 can be used to visualize the mapping of Mars. In Figure 16, the bands of latitudes to be mapped are shown in twenty-day intervals beginning after the final orbit trim and continuing throughout the nominal mission. In each of the four mapping sequences shown, 360 degrees of Mars longitude are covered between two parallels of latitude. As the mission progresses, these bands of latitude shift northward to follow the evening terminator and to fill in side-lap gaps between sequences of pictures taken on adjacent orbits. For an orbit inclination of 65 degrees and with an apsidal orientation angle

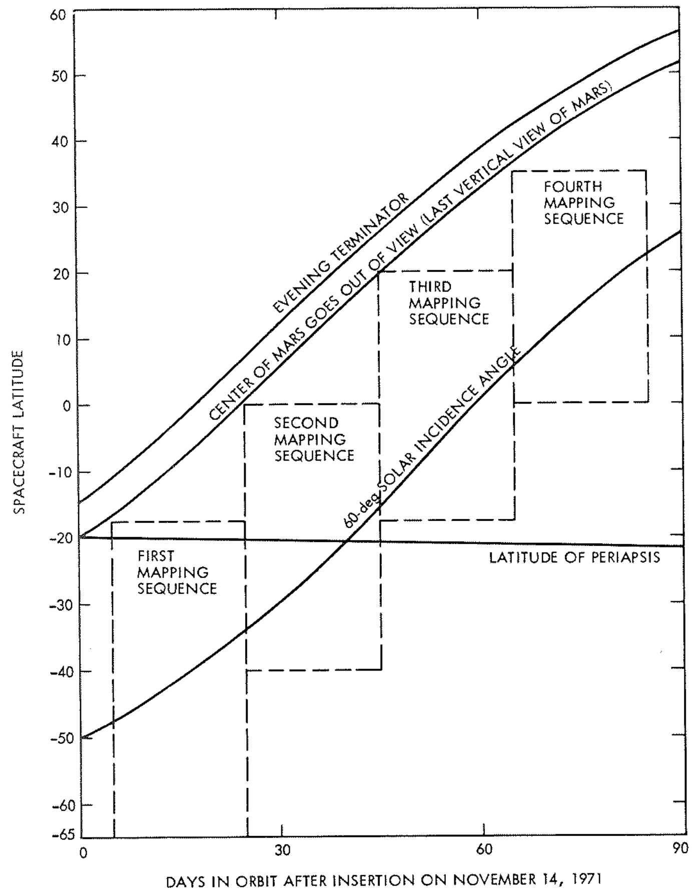


Figure 16. Mars Mapping Sequence

of 136 to 142 degrees, the latitude of periapsis is initially at about 15 to 20 degrees south latitude. As the mission progresses, the latitude of periapsis will decrease slightly because of a slight negative precession of the orbit line of apsides. When vertical wide-angle camera pictures are taken north or south of periapsis, the spacecraft altitude will be

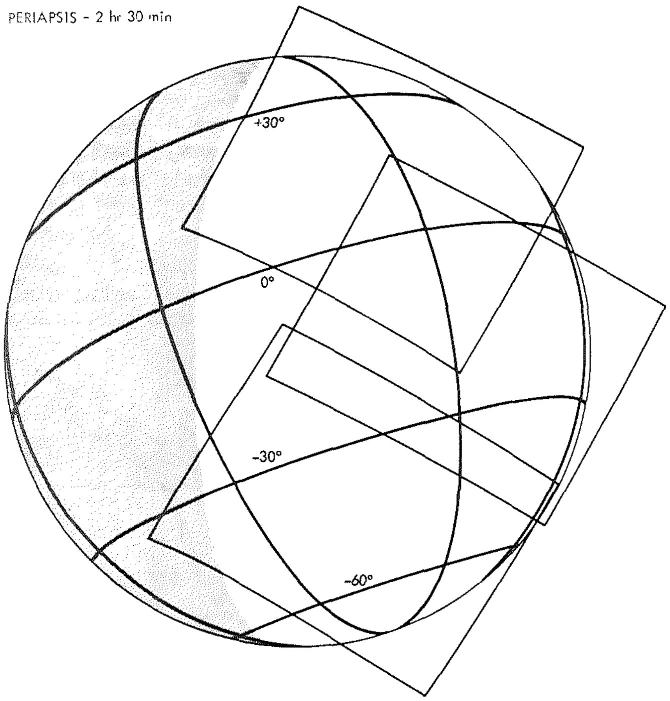


Figure 17. Typical Global Coverage Pictures

greater; the TV wide-angle pictures will have positive front-lap; the resolution will be reduced. Most of the wide-angle pictures will be taken vertically; the last vertical picture can be taken at a cone angle of 96 degrees. Late in the mission, the fourth mapping sequence could be extended northward to perhaps 45 degrees north latitude by fixing the scan platform at a cone angle of 96 degrees and at a clock angle of 305 degrees and then taking off-vertical wide-angle camera pictures up to the terminator. Thus, the planet can be mapped completely from 65 degrees south latitude to 45 degrees north latitude during the nominal mission. As can be seen in Figure 16, many pictures will be taken when the solar incidence angle exceeds 60 degrees.

When the spacecraft is in view of Goldstone, the planet is in view of the spacecraft scan platform, and the tape recorder is not being played back, the spectral data from the infrared radiometer, the infrared interferometer spectrometer, and the ultraviolet spectrometer are transmitted to the 64-m Goldstone station. Also, during the mapping sequence when the tape recorder is on, complementary multi-spectral data from the ultraviolet, visual, and infrared portions of the electromagnetic spectrum are obtained.

Shortly after the mapping sequence ends early in the mission near periapsis and the spacecraft enters Earth occultation, time communication with the Earth is temporarily lost. However, as the spacecraft enters and exits occultation, the S-band radio doppler data that is obtained will ultimately be used to determine the pressure profile of the Mars atmosphere. After Earth occultations cease, communication with the spacecraft during the 40 minutes following periapsis passage is maintained on Goldstone zenith passes. Between the time the mapping sequence ends and the limb of Mars passes out of view of the spacecraft scan platform, spectral data from the ultraviolet spectrometer, infrared, and infrared radiometer can be obtained from the night side of the planet beyond the evening terminator.

After the limb of the planet passes out of view of the scan platform, the ultraviolet spectrometer can be pointed to various regions of space surrounding Mars in search of Lyman alpha particles. The data will be transmitted in real time at 50 bits/sec. At this low data rate, continuous coverage can be provided independently from Goldstone by the world-wide network of 26-m (85-ft) diameter antennas.

The celestial mechanics experiment continues throughout the mission. However, the most useful data is obtained near each periapsis passage. One- and two-way doppler data are obtained on Goldstone zenith and nadir passes at 33-1/3 bits/sec, except for periods of Earth occultation. However, ranging data can be obtained only on Goldstone zenith passes.

After the spacecraft exits Earth occultation, playback of the tape-load of data obtained on the previous Goldstone zenith pass is completed in about three hours. After playback, Goldstone sets. When Goldstone is not in view, no tape recorder playbacks occur, and no high-rate spectral data is returned in real time.

During the period from 3 to 1.5 hours before periapsis, opportunities will exist to take TV wide-angle pictures of the lighted portion of the planet disc and of the atmosphere above it. This global TV coverage can be constructed from a mosaic of the wide-angle camera pictures taken on each nadir orbit. If three pictures are taken per orbit, a single wide-angle camera color filter may be used (see Figure 17). If six pictures are taken per orbit, two TV wide-angle camera color filters may be used, or some high resolution narrow-angle camera pictures of selected portions of the lighted limb and of the atmosphere above it can be taken. The best opportunities for global coverage occur early in the mission after the limb of Mars first comes into view of the scan platform (cone angle = 96 degrees and clock angle = 90 degrees). As the mission progresses, the limb and the center of Mars come into view when the spacecraft is closer to periapsis; the angular diameter of Mars is larger, and the number of TV wide-angle pictures required to mosaic the lighted portion of the planet escalates rapidly. Global coverage will probably end 25 to 30 days after insertion. The apsidal orientation angle, ψ, will be made as large as possible at orbit insertion to permit global coverage for the maximum practical period after arrival.

On the nadir pass, geodesy TV pictures will be taken about 1 hour and 30 minutes prior to periapsis, but they can not be played back at that time because Goldstone is not in view. South polar region TV or high-sun TV pictures will be taken on alternate nadir orbits. TV mapping, including recorded spectral data, will be obtained on each nadir pass.

The only real-time data that can be obtained on nadir orbits will be infrared radiometer data and ultraviolet spectrometer Lyman alpha data at 50 bits/sec or celestial mechanics one- and two-way doppler data at 33-1/3 bits/sec. This data will be received by the 26-m (85-ft) diameter antenna network.

Additional Science Sequences

After the spacecraft has been in orbit from 90 to 120 days, the north polar cap can be seen in the light by the scan platform of the spacecraft. This viewing period will occur from 0.5 to 1.5 hours after periapsis.

Photography of the Martian moons, Phobos and Deimos, will be obtained many times throughout the mission. The spacecraft will pass to within 5,000 to 10,000 km of the satellites at the optimum opportunities.

SPACECRAFT SYSTEM

The design of the Mariner Mars 1971 orbiter (see Figure 18) is essentially the same as that of Mariners 6 and 7 (Mariner Mars 1969), with some modifications made to meet the 1971 mission requirements (see Table 4). The most significant modification is the addition of a 1334-newton (300-lb) thrust rocket, which slows the spacecraft sufficiently near the planet to allow it to be captured in the gravitational field of Mars.

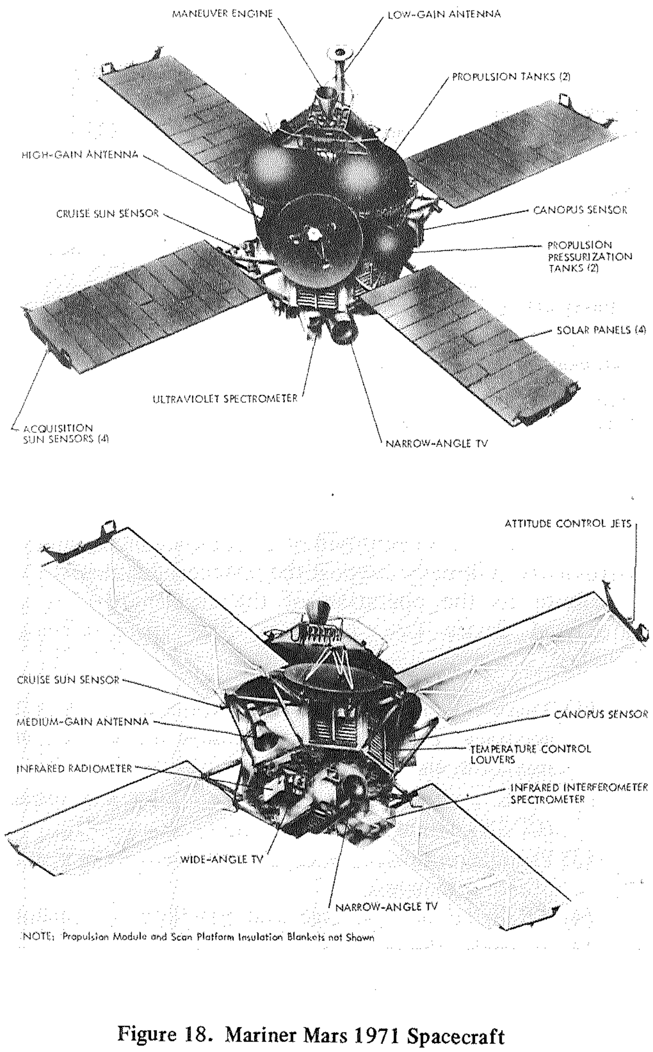


Figure 18. Mariner Mars 1971 Spacecraft

Table 4. Spacecraft Comparisons

Parameter	Mariner Mars 1964 Project	Mariner Mars 1969 Project	Mariner Mars 1971 Project
Spacecraft designation	Mariner 4	Mariners 6 and 7	Mariner 9
Total spacecraft weight (at launch)	261 kg	384 kg	1031 kg
Science instrument weight	23 kg	59 kg	68 kg
Propellant weight (at launch)	10 kg	10 kg	476 kg
Attitude control gas (N_2)	2.36 kg	2.45 kg	2.45 kg
Propellant pressurant gas (N_2)	N/A	N/A	14.5 kg
Computer memory capacity	No computer	128 words	512 words
Power supplied at Mars	194 W	380 W	450 W
Electrical part count (actual)	39,220	24,250	27,863 (approx)
Electrical part count (equivalent)	39,220	98,764	112,900 (approx)
Duration of near-Mars examination	30 min	30 min	90 days (2160 hours) minimum
Closest approach distance to Mars	9844 km	3379 km	1250 km

The spacecraft is composed of 19 subsystems; 4 are science instruments, 3 directly support the science subsystems, and 12 contribute to the operation of the spacecraft as a semi-automated device (see Table 5). The spacecraft design includes:

(1) Three-axis attitude control subsystem with a high-accuracy autopilot for midcourse maneuvers, orbital insertion, and orbital trims.

(2) Flight-programmable central computer and sequencer (CC&S) with a 512-word memory.

(3) Propulsion subsystem capable of performing a minimum of five maneuvers.

(4) All-digital data storage and handling equipment.

(5) Multiple channel telemetry subsystem with variable high-rate telemetry.

(6) Two-way communication and command capability based on the use of a low-gain; a medium gain; and a

two-position, high-gain antenna mounted to the spacecraft.

(7) Four solar power panels, one battery, and power conversion equipment.

(8) Temperature control equipment.

(9) A ground-commandable, two-degree-of-freedom scan platform for holding and pointing the science instruments.

(10) Planetary science instruments.

The spacecraft weighed about 1031 kg (2272 lb) at launch and measured 2.9 m (9.5 ft) to the top of the rocket motor and the low-gain antenna. With solar panels extended, the spacecraft spans 6.9 m (22 ft, 7.5 in.) across. Upon insertion into orbit around Mars, the spacecraft weighed about 590 kg (1300 lb) as a result of earlier separation of the adapter and the expenditure of fuel for trajectory-correction and orbital-insertion motor burns.

Structure

The basic octagon structure is an 18.2 kg (40-lb), eight-sided magnesium framework with eight electronics compartments encircling the spacecraft. These eight compartments contain the following equipment:

Bay I:	power regulator electronics assembly.
Bay II:	infrared interferometer spectrometer, planetary scan platform, and power conversion electronics assembly.
Bay III:	attitude control and CC&S electronics assembly.
Bay IV:	command and telemetry electronics assembly.
Bay V:	data storage electronics assembly.
Bay VI:	radio electronics assembly.
Bay VII:	data automation and television electronics assembly.
Bay VIII:	battery assembly.

The electronics assemblies fastened within the eight compartments also provide structural support to the spacecraft. This octagon provides the base for attaching the other spacecraft structures: propulsion support, adapter, antenna, solar panel, attitude control gas assemblies, and planetary scan platform. The outboard surface elements of the major electronics compartments are designed to function as thermal/shear plates when mechanically integrated to the spacecraft primary structure. Temperature transducers are installed in the compartment or on the shear plates of electronics compart-

Table 5. Spacecraft Subsystem Description

Subsystem	Functional Description	Weight, kg
Structure	Basic framework	131
Radio frequency	Communication to and from Earth	26
Flight command	Ground control of spacecraft operations	5
Power	Power for operation of spacecraft	75
Central computer and sequencer	Automatic timing and sequencing of spacecraft events	10
Flight telemetry	Organization and sequencing of data for transmittal to Earth	10
Attitude control	Guidance and attitude control	30
Pyrotechnics	Explosive latches, valves, and pinpullers	4
Cabling	Electrical connections between subsystems	35
Propulsion	Propulsion for trajectory corrections	574
Temperature control	Temperature control and maintenance of thermal environment	13
Mechanical devices	Mechanical damping, deployment, and latch mechanisms, scan platform	25
Data storage	Recording of data for playback to Earth at lower rates	11
Data automation	Control and synchronization of science instruments and data	6
Scan platform	Pointing control for scan platform holding science instruments	8
Ultraviolet spectrometer	Measurements of radiation intensity for selected bands of ultraviolet wavelengths	16
Television	Two television cameras for imaging pictures of Mars	26
Infrared radiometer	Measurements of temperatures on Martian surface	3
Infrared interferometer spectrometer	Measurements of infrared radiation intensity for selected bands of infrared wavelengths	23

ments occupying Bays I through VIII (see Figure 19). Six of the electronics compartments are temperature controlled by lightweight louver assemblies on the outer surfaces. Thermal shields cover most of the remaining area.

The spacecraft is linked to the Centaur stage of the launch vehicle by an adapter structure, which is bolted to the Centaur. A V-band clamp at the eight corners of the octagon

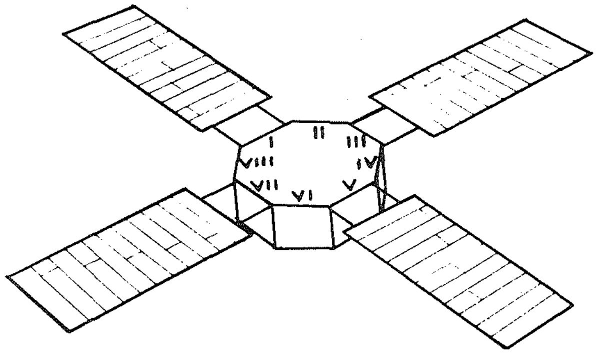


Figure 19. Spacecraft Electronics Compartments

holds the spacecraft during launch until the spacecraft is released from the adapter.

Radio

The radio-frequency subsystem receives commands and ranging signals transmitted by the deep space stations on Earth, and transmits science data, engineering data, and ranging signals back from the spacecraft.

Received and transmitted frequencies are in the S-band part of the spectrum. When no uplink signal is being received, the transmitted frequency is governed by an on-board auxiliary crystal oscillator. When the receiver detects and achieves phase lock with an uplink signal, the transmitted frequency is 240/221 times the received frequency. When the spacecraft-transmitted frequency is under control of the uplink received frequency, two-way doppler tracking is accomplished.

The telemetry data and ranging signal phase modulates the transmitter carrier. The spacecraft transmitter has two exciters and two traveling-wave-tube amplifiers (TWTA) to increase reliability. The TWTA output level is either low power (10 watts) or high power (20 watts).

Three S-band antennas (high, medium, and low gain) are used by the radio-frequency subsystem. The high-gain antenna structure consists of a reflector, which is an aluminum honeycomb parabola with a circular perimeter 1 m (40 in.) wide, and a feed supported at the focus of the parabola by a fiberglass truss. The antenna, which operates at a frequency of 2295 ± 5 MHz, is a two-position, pyrotechnic-activated device that allows optimum pointing of the antenna toward the Earth during the second half of the trajectory to Mars and during orbital operations.

The medium-gain antenna consists of a 10-cm (4-in.)-wide circular waveguide about 30 cm long, with a frustrum-shaped reflector about 24 cm wide mounted at its extremity. The antenna is coupled to the low-gain antenna and operates in the frequency range of 2115 to 2295 MHz. It is fixed to the spacecraft, so that it will point to the Earth when the spacecraft is oriented for the orbital insertion maneuver.

The low-gain antenna is a circular waveguide about 10 cm wide and 1.45 m long, with a frustrum-shaped reflector mounted at the extremity. This antenna is used to receive and transmit when the high-gain antenna cannot be used. It is mounted on the sunward side of the spacecraft and has a pattern almost symmetrical about the roll axis.

During periods when a ground station is transmitting the S-band uplink to the spacecraft, the spacecraft receiver, operating continuously throughout the mission, receives the uplink signal via the low- or medium-gain antenna. The signal may be the S-band carrier alone or the S-band carrier containing command and/or ranging information. The carrier containing command and/or ranging modulation is then processed and routed to the appropriate user location. The command data is sent to the flight command subsystem; the ranging data, when present in the uplink signal, is sent through the radio-frequency subsystem ranging channel to modulate the spacecraft transmitter in order to provide the turnaround ranging function. The ranging channel is turned off and on by ground command.

Flight Command

The spacecraft receives ground commands from Earth in addition to commands from the on-board CC&S. The radio-frequency subsystem receives these ground commands and relays them to the flight command subsystem, which detects and identifies the commands and issues them to the appropriate subsystem. Ground commands are required to execute trajectory corrections, to update functions related to spacecraft orbital operations, or to choose redundant elements in the event of certain component failures. Ground commands and the primary subsystem are listed in Table 6.

Power

The power subsystem provides a central supply of electrical power to operate the electrical equipment on the

Table 6. Ground Command List

Symbol	Name	Subsystem Destination
DC-1	Engineering Mode	FTS
DC-2	CC&S Readout Select (T)	FTS
DC-3	Playback Mode	FTS DSS
DC-4	DSS Ready Mode	DSS
DC-5	Engineering Data Rate Switch (T)	FTS
DC-6	FTS Redundant Elements Switch (T)	FTS
DC-7	Power Amplifier Switch (T)	RFS
DC-8	Exciter Switch (T)	RFS
DC-9	Ranging On/Off (T)	RFS
DC-10	Transmit Low	RFS
DC-11	Transmit High	RFS
DC-12	Adaptive Gate Step	A/C
DC-13	Maneuver Inhibit	A/C CC&S
DC-14	Maneuver Enable	A/C
DC-15	Canopus Gate Override	A/C
DC-16	DSS Record Mode	DSS
DC-17	Canopus Cone Angle Step	A/C
DC-18	Inertial Roll Control/Step	A/C
DC-19	Canopus Roll Control	A/C
DC-20	Roll Control Inhibit	A/C
DC-21	Roll Search/Step	A/C
DC-22	Select CW Reel Direction	DSS
DC-23	Select CCW Reel Direction	DSS
DC-24	Spare	PYRO
DC-25	RT Science No. 1 Mode	FTS
DC-26	Spare	PYRO
DC-27	Initiate Maneuver Sequence	CC&S
DC-28	Select Scan Stow Position	SCAN
DC-29	Accelerometer Scale Factor	CC&S
DC-30	Computer Inhibit	CC&S
DC-31	Computer Enable	CC&S
DC-32	Computer Maneuver Initiate	CC&S
DC-33	Tandem Maneuver	CC&S
DC-34	Scan On/Off (T)	PWR
DC-35	Select Variable Scan Reference	SCAN

Table 6. Ground Command List (Cont'd)

Symbol	Name	Subsystem Destination
DC-36	Initiate TV Mapping Sequence	DAS
DC-37	Boost Mode Enable/Inhibit (T)	PWR
DC-38	Battery Charger On/Off (T)	PWR
DC-39	DSS Slew Mode	DSS
DC-40	Gyros Inhibit	A/C
DC-41	Select Scan Cone Position	SCAN
DC-42	TWT High Power	RFS
DC-43	TWT Low Power	RFS
DC-44	RT Science No. 2 Mode	FTS
DC-45	Platform Unlatch	PYRO
DC-46	TV Cover Deploy	TV
DC-47	DSS On/Off (T)	PWR
DC-48	IRR Mirror Stow	IRR
DC-49	High-Gain Antenna Update	PYRO
DC-50	Battery Test Load On/Off (T)	PWR
DC-51	Disable Tolerance Detector	CC&S
DC-52	Computer Flag 7 Interrupt	CC&S
DC-53	Spare	PYRO
DC-54	Downlink On	PWR
DC-55	Downlink Off	PWR
DC-56	Select 16 kbps PLBK Rate	DSS
DC-57	Select 8 kbps PLBK Rate	DSS
DC-58	Select 4 kbps PLBK Rate	DSS
DC-59	Select 2 kbps PLBK Rate	DSS
DC-60	Select 1 kbps PLBK Rate	DSS
DC-61	Simulate Sun Gate	A/C
DC-62	Select Pre-Aim Backup Mode	A/C
DC-63	Roll Gyro On	A/C
DC-64	Switch Pre-Aim Backup Bias	A/C
DC-65	Open Press, P1/Prop. O1, F1	PYRO
DC-66	Close Pressurant P2	PYRO
DC-67	Close Propellant O2, F2	PYRO
DC-68	Open Pressurant P3	PYRO
DC-69	Open Propellant O3, F3	
DC-70	Close Pressurant P4	PYRO
DC-71	Close Propellant O4, F4	PYRO
DC-72	Open Pressurant P5	PYRO

Table 6. Ground Command List (Cont'd)

Symbol	Name	Subsystem Destination
DC-73	Open Propellant O5, F5	PYRO
DC-74	Charger Auto. Switchover On/Off (T)	PWR
DC-75	Propulsion Heater On/Off (T)	PWR
DC-76	DAS On	PWR
DC-77	UVS and IRR On/Off(T)	PWR
DC-78	TV On/Off(T)	PWR
DC-79	IRIS On/Off(T)	PWR
DC-80	Science Off	PWR
DC-81	Select Battery Charge Rate (T)	PWR
DC-82	Spare	DAS
DC-83	Switch RTS No. 2 Data Rate (T)	DAS
DC-84	Computer Flag 6 Interrupt	CC&S
DC-85	Enable Tolerance Detector	CC&S
DC-86	Computer Flag 8 Interrupt	CC&S
QUANTITATIVE COMMANDS		
QC-1	Platform Cone Slew Positive	SCAN
QC-2	Platform Cone Slew Negative	SCAN
QC-3	Platform Clock Slew Positive	SCAN
QC-4	Platform Clock Slew Negative	SCAN
CODED COMMANDS		
CC-1	Computer Load	CC&S
CC-2	Computer Load	CC&S
CC-3	Word Interrogate	CC&S
CC-4	Sequencer Load	CC&S
CC-20	DAS Coded Command	DAS

spacecraft. It also provides the required switching and control functions for the effective management and distribution of that power, as well as a central timing function for the spacecraft. The power, which is derived from four photovoltaic solar panels and a rechargeable battery, is converted and distributed in the following forms:

(1) 2.4-kHz (kilocycles per second), single-phase, square-wave power for engineering and science subsystems, and for the propulsion module and cone actuator heaters as required.

(2) 400-Hz, three-phase, quasi-square-wave power to the attitude control subsystem for gyro motors.

(3) 400-Hz, single-phase, square-wave power to the scan platform.
(4) Regulated 30-V DC power for the engine valve and gimbal actuators.
(5) Unregulated DC power to the battery charger, heaters, and radio-frequency subsystem (RFS) for the TWTA power supply.

The four solar panels used for mounting of the solar cells provide a total area of 7.7 square meters. Each panel is 2.14 m long by 0.90 m wide. The cell surface substrate is a single skin on transverse corrugations supported by two cross-braced longitudinal spars. The panels are supported during launch in a 15-degree-from-vertical position (see Figure 9). They are opened after spacecraft separation by pinpullers at one end of each boost damper pair and are deployed about 105 degrees by a deployment mechanism. After deployment, the panels are latched in a plane normal to the spacecraft centerline (as in Figure 7) by engaging the attached damper mechanism.

During the period from launch to Sun acquisition, the 400-Hz, three-phase inverter; the main 2.4-kHz inverter; and the 30-V DC regulator are energized. Unregulated DC power is supplied to the RF power supply and the DC heaters. Battery charger and battery booster are on, or in the enabled mode. During this phase of flight, the raw power is supplied from the spacecraft nickel-cadmium battery, which has a minimum capacity of 20 ampere-hours.

Raw electrical power is supplied from the solar panels during interplanetary cruise. A part of the solar panel output is furnished directly to the RF subsystem for the traveling-wave-tube power supply and also to DC heaters and the battery charger. All other cruise loads are supplied from the 2.4-kHz square-wave supply. The 400-Hz, single-phase inverter remains off and the 400-Hz, three-phase inverter is turned off. Battery charger and battery booster remain in the enabled mode. The battery is kept fully charged.

During the maneuvers, the 400-Hz, three-phase supply; the 2.4-kHz power to attitude control gyro electronics; and the regulated DC power for the engine valve and gimbal actuators are turned on in response to signals from the attitude control subsystem. The raw power is supplied from solar panels or from the spacecraft battery, or from both in a load-sharing mode, depending on the solar panel power available.

The outputs of the power subsystem during orbital insertion are essentially the same as during the maneuvers. During the orbital insertion yaw turn, the spacecraft power source is transferred from the solar panels to the battery because of the magnitude of the turn, and remains on the battery until the Sun is reacquired after completion of the turn. At this time, about 350 watt-hours are required from the battery. The outputs of the power subsystem are the same during the orbital trims as during the maneuvers.

The outputs of the power subsystem during orbital cruise are essentially the same as in interplanetary cruise, except that the battery charger is in the high-rate mode to recharge the battery after orbital insertion or orbital trims. Approximately 95 watts of raw power are required for battery charging. The battery charger is switched to low rate when the battery is fully charged.

During orbital operations, raw electrical power is supplied from the solar panels, as required. The main single-phase outputs are switched to provide power for the science instruments, data automation subsystem, data storage subsystem, and scan platform. The 400-Hz, three-phase inverter is turned on when necessary to supply power to the roll gyro.

Central Computer and Sequencer

The central computer and sequencer (CC&S) subsystem provides timing and sequencing services for the other spacecraft subsystems. The sequencing is generated by a special-purpose computer with fixed sequencer redundancy in the maneuver mode. Timing and sequencing (except the fixed sequencer) are programmed into the CC&S before launch and can be modified during flight by coded command (CC).

Trajectory-correction maneuvers are, in the normal operating mode, fixed in sequence with roll and yaw directions; spacecraft velocity increments are variable by coded command. The normal operating mode, defined as the tandem mode, operates the computer part of the CC&S concurrently with the fixed sequencer and requires that events coincide between each part. Either the computer or fixed sequencer may execute the maneuver independently if so directed by direct command (DC).

The CC&S is capable of sending the commands listed in Table 7. Timed events are initiated in six basic sequences:

(1) **Launch**. Sequence starts with the loading of the CC&S program before liftoff, and ends when the spacecraft becomes fully stabilized in flight. It is desirable that the first event after launch be programmed to occur 1 hour or more after CC&S inhibit release. Launch events are normally programmed with minutes resolution after the first event.

(2) **Cruise**. Sequence starts at the same time as the launch sequence and lasts for the mission duration. Launch, maneuver, and orbital sequences are essentially superimposed on the cruise sequence. Cruise events are normally programmed with hours resolution.

(3) **Maneuver**. Sequence starts by DC command or by

computer event 5A (see Table 7). Three modes of maneuver sequencing are possible; the tandem mode (normal), the fixed sequencer mode, and the computer mode. The maneuver sequence is programmed for seconds resolution between events. The fixed sequencer maneuver can be interrupted by computer event 5B (see Table 7).

(4) **Pre-orbital insertion.** Sequence is comprised of science data acquisition beginning with science instrument turn-on several days before insertion into Mars orbit.

(5) **Orbital insertion.** Sequence starts by DC command or by computer event 5A. The orbit-insertion sequence is a hybrid mode where the computer and sequencer operate in parallel. The sequence requires both minutes and seconds resolution between commands.

(6) **Orbit.** Sequence starts after a correct orbit is attained and continues for the life of the mission, or until changed by coded command. The sequence requires hours, minutes, and seconds resolution.

Table 7. CC&S Command List

Symbol	Name	Destination
2A	Test Radio	RFS
2B	Transmit Low	RFS
2C	TWT Low Power	RFS
2D	TWT High Power	RFS
2E	Transmit High	RFS
4A	Select Battery Charge Rate (T)	PWR
4B	Battery Charger On/Off (T)	PWR
4C	DAS On	PWR
4D	UVS and IRR On/Off (T)	PWR
4E	TV On/Off (T)	PWR
4F	IRIS On/Off (T)	PWR
4H	DSS On/Off (T)	PWR
4J	Scan On/Off (T)	PWR
4K	DAS/TV 2.4 kHz off	PWR
4L	Propulsion Heater On/Off (T)	PWR
4M	Downlink On	PWR
4N	Downlink Off	PWR
5A	Initiate Maneuver Sequence	CC&S
5B	Sequencer Maneuver Interrupt	CC&S
5C	CC&S B Frame Start Enable	CC&S

Table 7. CC&S Command List (Cont'd)

Symbol	Name	Destination
6A	Engineering Mode	FTS
6B	Engineering Data Rate Switch (8-1/3)	FTS
6C	Engineering Data Rate Switch (33-1/3)	FTS
6D	RT Science No. 1 Mode	FTS
6E	RT Science No. 2 Mode	FTS
7A	A/C On	A/C
7B	Canopus Sensor On	A/C
7C	Adaptive Gate Step	A/C
7D	Canopus Cone Angle Step	A/C
7E	Autopilot On	A/C
7F	CC&S Stray Light Signal	A/C
7G	Gyros Inhibit	A/C
7M1	Gyros On	A/C
7M2	All Axes Inertial	A/C
7M3	Turn Polarity Set	A/C
7M4	Roll Turn	A/C
7M5	Yaw Turn	A/C
7M6	A/C Maneuver Mode	A/C
8A	Deploy Solar Panels	PYRO
8B	Spare	PYRO
8C	Platform Unlatch	PYRO
8D	High-Gain Antenna Update	PYRO
8E	Open Press, P1/Propellant	PYRO
8F	Close Pressurant P2	PYRO
8G	Close Propellant O2, F2	PYRO
8H	Open Pressurant P3	PYRO
8J	Open Propellant O3, F3	PYRO
8K	Close Pressurant P4	PYRO
8L	Close Propellant O4, F4	PYRO
8N	Open Pressurant P5	PYRO
8P	Open Propellant O5, F5	PYRO
8M6	Open/Close Engine Valve	PYRO
16A	Select 16 kbps PLBK Rate	DSS
16B	Select 8 kbps PLBK Rate	DSS
16C	Select 4 kbps PLBK Rate	DSS
16D	Select 2 kbps PLBK Rate	DSS
16E	Select 1 kbps PLBK Rate	DSS

Table 7. CC&S Command List (Cont'd)

Symbol	Name	Destination
16F	DSS Ready Mode	DSS
16G	DSS Record Mode	DSS
16H	Playback Mode	DSS
16J	Advance to Track 1 LEOT	DSS
20B	Switch RTS No. 2 Data Rate (T)	DAS
20C	Initiate TV Mapping Sequence	DAS
20D	Take TV Picture Pair	DAS
20G	Reset DAS Orbit Logic	DAS
20H	IRIS IMCC Mirror Enable	DAS
20J	TV Beam Current On/Off	DAS
31A	Platform Clock Slew Positive	SCAN
31B	Platform Clock Slew Negative	SCAN
31C	Platform Cone Slew Positive	SCAN
31D	Platform Cone Slew Negative	SCAN
31E	Select Scan Stow Position	SCAN
31F	Select Scan Cone Position	SCAN
31G	Select Variable Scan Reference	SCAN
36A	TV Cover Deploy	TV
38A	IRR Mirror Stow	IRR

Flight Telemetry

The flight telemetry subsystem, by suitable modulation of the radio subsystem RF signal, enables the formatting and transmitting of data on three channels: one for science, one for engineering, and one for high-rate data. This subsystem also provides data rate, data mode, and modulation index switching.

Attitude Control

The attitude control (A/C) subsystem provides continuous spacecraft flight stabilization and orientation after separation from the launch vehicle. It automatically orients the spacecraft with respect to the lines of sight to the Sun and the star Canopus. Upon receipt of commands from the CC&S, the A/C subsystem orients the spacecraft to align the propulsion subsystem thrust axis in the direction commanded for the trajectory-correction maneuver, orbital insertion maneuver, or orbital trim maneuver. During maneuvers, the A/C subsystem maintains spacecraft orientation and stability in pitch and yaw by two-axis gimbal control of the rocket engine and in roll by the attitude control jets (small gas jets mounted on the ends of the solar panels). An accelerometer signal is provided to the sequencer in the CC&S for control of the velocity magnitude. At the end of a maneuver sequence, a signal from the CC&S initiates attitude control reorientation of the spacecraft to the Sun and Canopus.

Before and during launch, all major components of the A/C subsystem are disabled except the gyros, accelerometer, and gimbal servos, which are enabled to withstand the launch and separation environment.

At separation from the launch vehicle, the A/C subsystem is enabled by a signal from a separation-actuated switch, which is backed by redundant CC&S command. The A/C subsystem reduces the initial rates to an acceptable rate deadband and then orients the spacecraft to the Sun. If the Sun enters the Canopus sensor's field of view during launch and Sun acquisition, a Sun shutter is actuated to protect the sensor. Power to the Sun shutter circuitry is inhibited at Sun acquisition by the Sun gate signal.

Upon completion of Sun orientation, the A/C subsystem maintains the roll rate within the rate deadband until the command to inititate Canopus acquisition is issued by the CC&S. Upon receipt of the command, the Canopus sensor is energized and its output is fed into the roll switching amplifier, causing the spacecraft to roll counterclockwise at a controlled rate in search of the roll-reference star. Stars other than Canopus are rejected on the basis of cone angle and brightness settings. The cone angle of a celestial object is defined as the angle from the spacecraft-object line (see Figure 20). The clock angle of a celestial body is defined as the angle between a plane containing the Sun, spacecraft, and Canopus and a plane containing the Sun, spacecraft, and object. The angle is measured from the Sun-spacecraft-Canopus plane and defined as positive, in the clockwise direction, when looking toward the Sun from the spacecraft.

Cone angle discrimination is obtained by limiting the sensor cone angle field of view to ± 5.5 degrees from nominal. In order to accommodate variations of Canopus cone angle, the Canopus sensor cone angle field of view is incremented periodically by commands from the CC&S. When a star satisfying the brightness gate logic is detected, the A/C subsystem ends the roll search. The detection of Canopus also initiates a fixed interval timer (180 seconds usually) which turns off the gyros.

Completion of Sun and Canopus reference acquisition initiates the cruise mode of operation. In this mode, rate damping is achieved by the use of derived rate networks. The A/C subsystem is capable of automatic reacquisition of Sun and Canopus references in the event that either or both of these

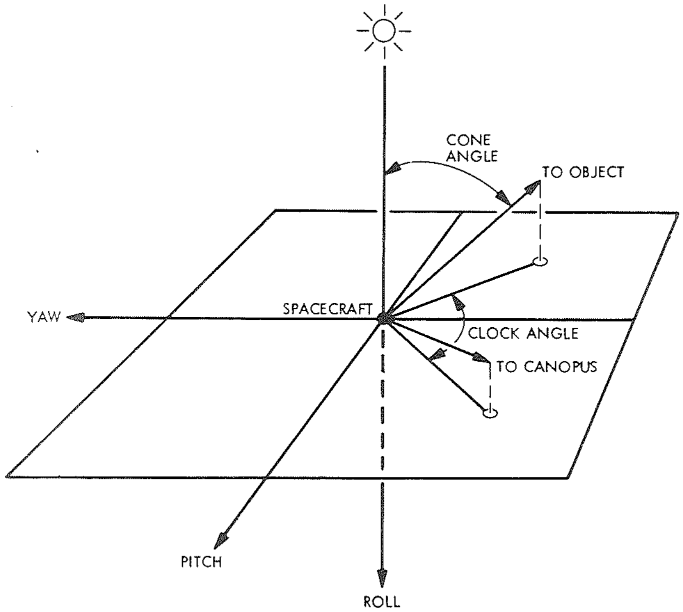


Figure 20. Cone and Clock Angles

references is lost because of an external disturbance.

Before the commanded turns for a specific maneuver, the engine thrust vector is pointed through the spacecraft center of mass by the pre-aim circuits. The pre-aim circuits function by accepting from the CC&S a digital word that represents the required gimbal actuator extension, and by appropriately biasing the null extension of the actuators.

In response to commands from the CC&S, the A/C subsystem orients the spacecraft to any commanded attitude for trajectory correction, orbital insertion, or trim maneuvers. This orientation is accomplished by a roll/yaw sequence of time-controlled turns. The turn rate is constant, and the turn magnitude is controlled by the duration of the turn command. The spacecraft attitude is controlled by the signals from three single-axis gyros.

At engine ignition, several changes take place in the attitude control subsystem. The pitch and yaw switching amplifiers are disabled; path guidance circuits, in the autopilot, are enabled. Rate and position signals for autopilot control of the spacecraft are obtained from the gyros and associated electronics. Control torques are generated in pitch and yaw by gimballing the rocket engine assembly and in roll by the A/C jets. At the end of the engine burn, the path guidance is disabled and the switching amplifiers are enabled. The effects of the engine shutdown transient is controlled by the A/C jets.

An accelerometer signal is sent to the CC&S to provide closed-loop control of the maneuver velocity increment.

Upon completion of the maneuver sequence and in response to commands from the CC&S, the A/C subsystem reorients the spacecraft by unwinding the commanded turns of the maneuver. At the completion of the unwinding sequence, the CC&S commands the A/C subsystem to initiate Sun acquisition. When the Sun is acquired, as signalled by the Sun gate, the A/C subsystem initiates acquisition of the star Canopus.

The orbital insertion maneuver is commanded and executed in the same way as the trajectory-correction maneuver.

The orbital cruise mode of operation is similar to the transit cruise mode previously described except that a CC&S generated signal is used to simulate a stray-light signal. The stray-light signal coincides with conditions where lighted crescents of Mars enter the Canopus sensor stray-light field of view, causing the Canopus sensor signals to be erroneous. The mode of operation during these stray-light conditions is one in which the A/C subsystem logic, upon receiving the stray-light signal, turns on the roll gyro and starts the fixed-interval (usually 180 seconds) timer. At the end of the fixed interval, the inertial reference unit is commanded into the integrate mode, and the Canopus sensor signal is replaced with the position signal from the inertial reference unit. At the end of the stray-light signal, the Canopus sensor is commanded to reacquire Canopus.

Orbital trim maneuvers are commanded and executed as described for the trajectory-correction maneuvers. Required pre-aim for orbital trims differs from that of previous maneuvers because of the large transverse center-of-mass shift with orbital insertion fuel depletion.

Pyrotechnics

Electrically initiated explosive devices are used for spacecraft separation, solar panel release, high-gain antenna position change, scan platform release, and propulsion system valve actuation. Functions can be initiated by either direct command to the spacecraft or by commands stored in the spacecraft central computer and sequencer. Pyrotechnic firing is accomplished by capacitor discharge into the intended device.

Mechanical Devices

The devices used in this subsystem are associated with latching, structural damping, non-servo-controlled actuation, planetary experiment support, and separation-activated switching devices that are used in the mechanical design of the spacecraft. Mechanical devices include:

(1) Solar panel dampers.
(2) Solar panel deployment and latch mechanisms, including the switch assemblies for indications of deployment of panels.
(3) High-gain antenna deployment mechanism.
(4) Planetary scan platform.
(5) Pyrotechnic arming switch.
(6) Spacecraft-initiated timer.
(7) Spacecraft-separation mechanisms.
(8) Spacecraft V-band clamp (separation for Centaur adapter).
(9) Medium-gain antenna energy attenuation plug.

Propulsion

The function of the propulsion subsystem is to provide directed impulse, upon command, to accomplish in-transit trajectory corrections, an orbital insertion maneuver at encounter to transfer from a flyby to an orbiting trajectory about the planet Mars, and subsequent trim maneuver.

This storable bipropellant propulsion subsystem is an integrated, pressure-fed, multi-start, fixed-thrust subsystem that uses nitrogen tetroxide (N_2O_4) and monomethylhydrazine (MMH) as propellants. The primary subassemblies are a dual-tank nitrogen reservoir, a pressurant control assembly that provides pressurant isolation and regulation, two check and relief valve assemblies, two propellant tanks with positive expulsion bladders, two propellant isolation assemblies, a gimballed 1334-newton (300-lbf)-thrust rocket engine assembly with an electrically operated bipropellant valve, and the propulsion module structure. The rocket engine contains a thick Beryllium combustion chamber which conducts heat rapidly and is cooled by fuel sprayed on the inside walls. The nozzle is made of high temperature steel and is red hot during firings.

The subsystem is pressurized by gaseous nitrogen from high pressure storage tanks. Welded or brazed tubing and components connections are used. Metal seals are used to minimize the effects of irradiation, hard vacuum, temperature, and long-term storage on critical subsystem joints. Multiple pyrotechnic valves, arranged in three groups with normally open and normally closed valve branches, provide the capability to isolate propellant and pressurant for the long periods of space storage. The subsystem is capable of being fueled, pressurized, and monitored before installation on the spacecraft.

At launch, the propellants and high-pressure gas supply are isolated by the pyrotechnic valve assemblies. Before the first trajectory correction, the engine valve must be opened to bleed the air trapped between the normally closed propellant pyro valves and the engine valve. Actuation of valves allows the propellant tanks to go to operating pressure, and propellant flows down to the engine valve. The trajectory-correction maneuver is then performed by opening the engine valve, causing the propellants to flow into the thrust chamber, undergo hypergolic ignition, and continue to burn until such time as the desired velocity increment is obtained. At this time, the engine valve is closed by removing its electrical power. The propellant and pressurant lines are then closed to guard against leakage when tracking data confirm that no more propulsion maneuvers will be required before the nominal time of the second trajectory correction.

The pressurant and propellant lines are reopened just before the second trajectory correction. These lines will remain open through the second midcourse firing, which is expected to be only 1 or 2 seconds long, the 15-min orbit insertion firing, and two short orbit trim firings to place the spacecraft precisely into the desired orbit.

After tracking data confirm correct orbital characteristics, the propulsion fluids are isolated by the operation of valves for the rest of the mission.

Commands for event sequencing originate from the CC&S and/or the flight command subsystem. Actuation of pyro valves and management of solenoid power is accomplished by power switching in the pyrotechnics subsystem. Thrust vector control is provided by the A/C subsystem through the use of gimbal actuators for pitch and yaw control and cold-gas jets for roll control.

Temperature Control

Potentially damaging environmental conditions of space and heat generated by the spacecraft require that the temperatures of the spacecraft subsystems be controlled in order for all equipment to function correctly. The four major variables that affect the temperature of spacecraft components are incident solar radiation, electrical power dissipation, thermal transfer between components, and thermal radiation of the spacecraft into space. Various passive (shields, thermal blankets, paint, polished surfaces) and active devices (variable-emittance louver assemblies) are used to achieve temperature control.

Multi-layer thermal blankets are employed on the sunlit (top and bottom) sides of the spacecraft. Both blankets are lightweight, thermal boundaries. The purpose of the top blanket is to isolate the propulsion module and bus from the Sun; the bottom blankets minimize thermal gradients within the bus and scan platform and force the internally dissipated power to be rejected to space throughout the louvered bay faces.

Temperature-control louvers are installed on all spacecraft bays except Bays IV, and VI. Bay IV is covered with a polished, low-emittance aluminum shield, and Bay VI is covered with high-emittance white paint.

Data Storage

At many times during the mission, the spacecraft acquires data faster than the data can be transmitted to Earth. The data storage subsystem stores the data on a digital tape recorder until it can be transmitted to Earth at a slower rate.

This subsystem records data supplied by the data automation subsystem in the form of a serial stream of pulses. The data, recorded at a rate of 132.2 kbps, consists of digitized video from the television subsystem formatted with data from the other science instruments. Recording is automatically stopped: (1) when the tape recorder is filled, (2) by command from the CC&S, or (3) by ground command. When the ground antennas are ready to accept the data, the data are played back through the flight telemetry subsystem to the radio-frequency subsystem at a slower rate than recorded. Five playback data rates (16.2, 8.1, 4.05, 2.025, and 1.0125 kbps) are available and selectable by commands from the flight command subsystem or the CC&S.

Data Automation

The data automation subsystem acts as the signal interface between the science instruments and all other subsystems of the spacecraft. This subsystem:

(1) Controls and synchronizes the science instruments within a fixed timing and format structure so that the instrument internal sequencing is known, and sends commands to the instruments as required.

(2) Provides the necessary sampling rates, both simultaneous and sequential, to ensure meaningful science data.

(3) Performs the necessary conversions and encoding of the several forms of science data, and places them in a suitable format.

(4) Buffers the science data and sends it either to the flight telemetry subsystem at 50 bps or to the data storage subsystem at 132.2 kbps, as appropriate.

(5) Issues and receives commands that pertain to the operation of the science instruments to and from other spacecraft subsystems.

Planetary Scan Platform

The scan control subsystem provides precise angular pointing control of the two-degree-of-freedom (clock and cone axes) gimballed support structure or platform, upon which the planet-oriented science instruments are mounted. The instruments mounted on the scan platform are:

(1) Wide- and narrow-angle television cameras.
(2) Ultraviolet spectrometer.
(3) Infrared interferometer spectrometer.
(4) Infrared radiometer.

At launch, the scan platform is secured in the stowed position. One day after launch, a direct ground command or CC&S event signals the pyrotechnics subsystem to unlatch the scan platform.

The scan platform operates in the following modes: pre-orbital television, orbital science, and orbital cruise. In the pre-orbital television mode, the platform is moved so that a series of television pictures can be taken of the planet. In the orbital science mode, the platform is stepped sequentially through a series of pointing directions. The scan pointing positions are programmed in flight, and during the orbital sequence, by CC&S commands or radio quantitative commands. In the orbital cruise mode, the scan subsystem is turned off and the platform remains in the orientation of the preceding television picture.

Four reference potentiometers control the clock and cone angles for the start of the pre-orbital and orbital science sequences. The reference potentiometers are coupled through a gear train to four-step motors. Two identical clock and cone sequencing circuits supply pulses to turn the energized step motor. In a typical scan operation, the sequencing circuits receive either clockwise or counterclockwise pulses, spaced 1 second apart, from either the flight command subsystem or CC&S.

The scan platform can be pointed to within 1/2 degree of a desired direction; after moving the platform to the desired position, the actual direction will be known to within 1/4 degree in both cone and clock angles.

MISSION OPERATIONS

The Mission Operations System (MOS) has the primary responsibility for operating and controlling the two spacecraft in flight. The Tracking and Data System (TDS) provides tracking of the spacecraft and provides two way communications between the spacecraft and the JPL Space Flight Operations Facility (SFOF). Together, TDS and MOS process, record, and display all engineering and scientific data obtained from the spacecraft.

The quantities of tracking, telemetry, and operational data produced by any mission must be processed and displayed quickly and reliably. Tracking data must be evaluated immediately as the location; velocity, and direction of the spacecraft must be known at all times; in case of a spacecraft problem, a correction must be decided immediately.

Orbital

Because Mariner Mars 1971 will provide the first opportunity to acquire and analyze information regarding the planet while the spacecraft continues to obtain data, greater demands obviously will be placed upon the facilities, computer software programs, and supporting personnel than for any previous mission. To meet these demands, the MOS organization (see Figure 21) will be divided into two major levels: planning and analysis, and operations execution. Operations planning and

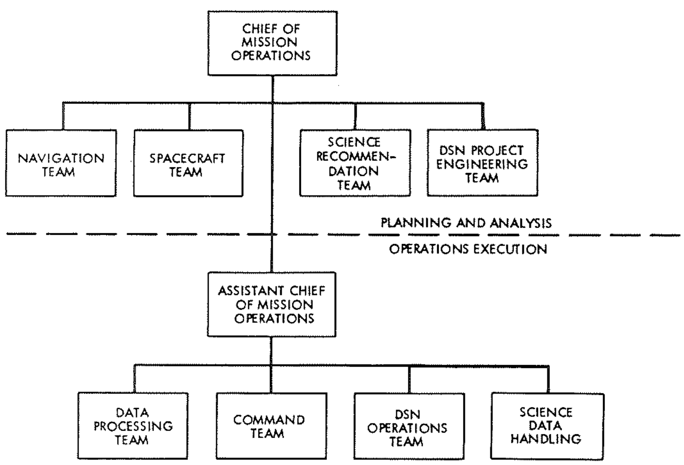


Figure 21. Organization of Mission Operations System for Mariner Mars 1971

analysis activities require the knowledge of navigation, of the spacecraft, and of the experiments. Planning and analyses will be conducted for 8 to 10 hours a day, under the direction of the Chief of Mission Operations (CMO), by four teams:

(1) Navigation Team, for spacecraft navigation and scan platform geometry analysis.

(2) Spacecraft Team, for Spacecraft System performance evaluation and prediction through engineering telemetry analysis, and development of alternatives in using the spacecraft to satisfy mission requirements.

(3) Science Recommendations Team, for analysis of science data and recommendations regarding science operations plans and priorities.

(4) Deep Space Network (DSN) Engineering Team, for allocation of DSN resources, operations planning, and configuration control.

Operations execution functions will be conducted 24 hours a day for 90 days, under the direction of an Assistant CMO, by four teams:

(1) Science Data Handling Team, for collection and coordination of science data processing requirements, assembly and dissemination of data from a science data library, and for producing the science experiment records for which the MOS is responsible.

(2) Data Processing Team, for planning, scheduling, and coordinating all data processing; and for furnishing and scheduling personnel to operate Project-supplied hardware and computer programs.

(3) Command Team, for continuously ensuring spacecraft utilization according to plans specified by the CMO; for translating mission operations plans into command sequences and initiating their transmission to the spacecraft; for guarding the spacecraft against incipient failure, to the degree possible; and, if necessary, for requesting additional resources to analyze spacecraft or science instrument anomalies.

(4) DSN Mission Operations Team, for operation of the DSN and coordination of tracking support.

Planning and operations functions for the orbiting spacecraft will be conducted simultaneously. That is, daily operational decisions for the spacecraft will be made at contiguous planning sessions by the planning and analysis teams. The functions of the teams are keyed to data receipt at Goldstone, the primary station for receiving high-rate science telemetry. The 24-hour operations cycle will start at Goldstone rise (when Goldstone 64-m antenna rises above the horizon sufficiently to establish radio communications with the spacecraft); data will be received and processed until Goldstone set. While data processing continues, data analyzing will be in process. Planning for the next 24-hour cycle will begin after the first part of the data received from Goldstone has been analyzed.

A daily Science Recommendations Team meeting will be held to review science operations plans and, based on analysis of data acquired, to suggest changes to operations plans. The recommendations of the team will be presented at the daily planning meeting, at which operations plans for the immediate 24-hour cycle are approved, and plans for the next 24-hour cycle are formulated. The CMO presides at the meeting, with active participation by the Assistant CMO and members of the planning and analysis teams. All recommendations, either long- or short-term, are made to the CMO at this time. This meeting is the end of the day's activity.

After the meeting, the Adaptive Mode Planning System (AMPS), a set of computer programs, will be initiated for the spacecraft. The AMPS output will result in a command file for each spacecraft and a printed Mission Sequence of Events (combined sequence of commands to be executed by the spacecraft and supporting ground activities) to support the day's plans.

At the completion of the AMPS run, another run will be initiated to produce a forecast of the next day's events. The forecast is the basis for planning and analysis activities, ending in the next daily planning meeting.

Although some variations in this cycle will occur because of special events, the daily operations profile described here and shown in Figure 22 will be used.

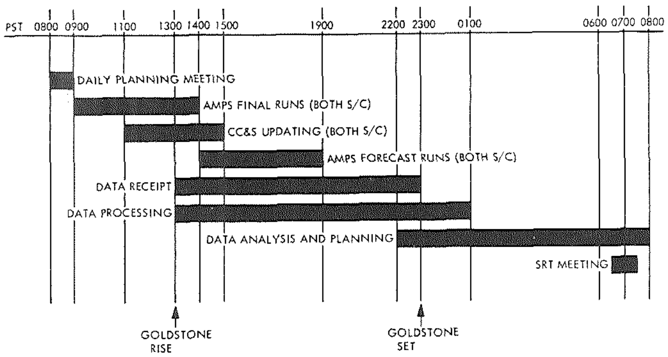


Figure 22. Profile of Daily Orbital Operations for Mariner Mars 1971 Spacecraft (S/C). Pacific Standard Times are Approximate

Figure 23 shows the Mariner Mars 1971 operations area of the Space Flight Operations Facility (SFOF) at the Jet Propulsion Laboratory in Pasadena. The positions of the Operations Control Team are at the round console in the middle of the figure. The adjacent areas occupied by the eight teams shown in Figure 21 are not included in this figure.

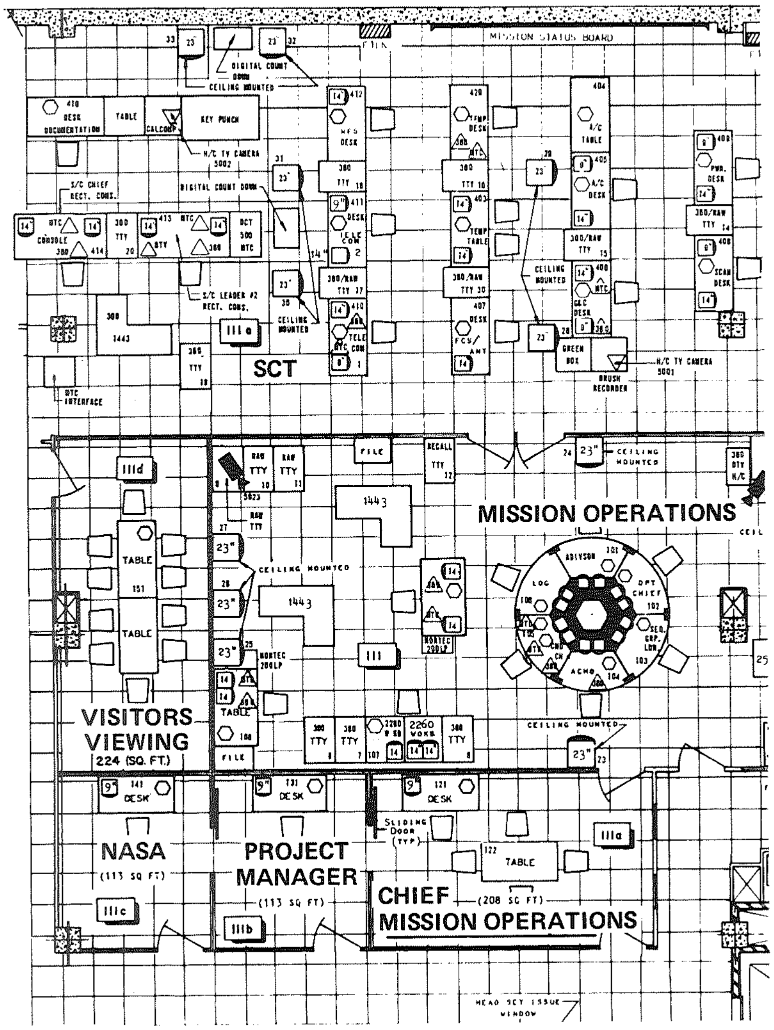
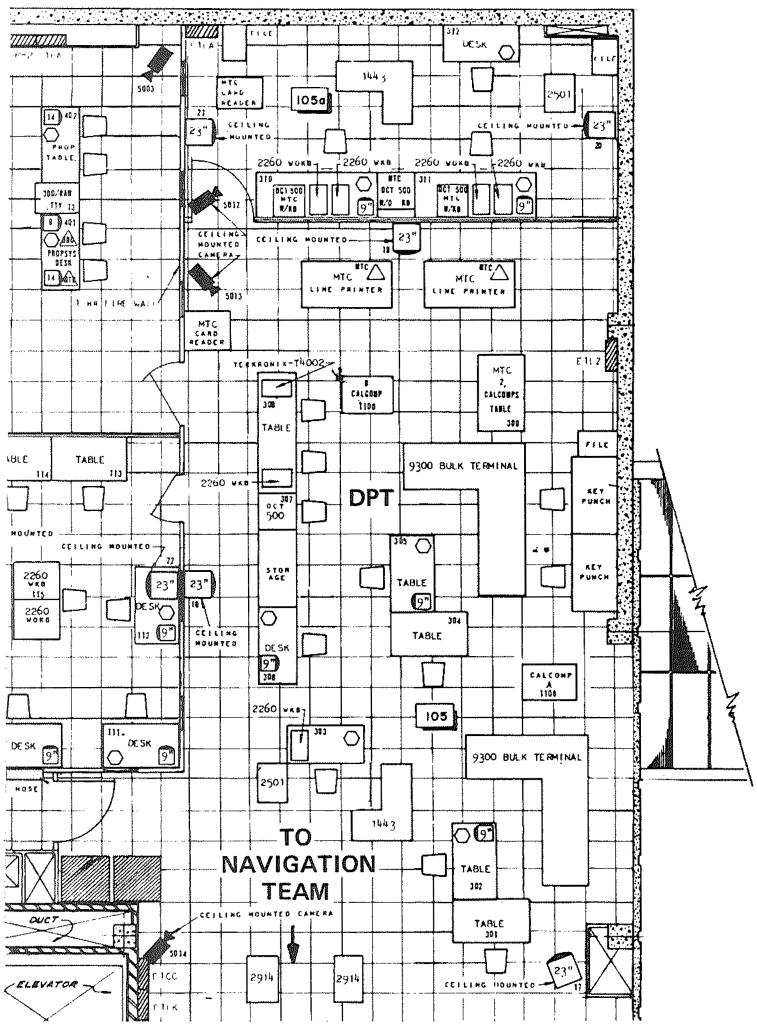


Figure 23. Mariner Mars 1971 Operations Area of SFOF

The DSN will support orbital operations by acquiring, and immediately routing to the SFOF, data telemetered from the spacecraft at rates up to 16.2 kilobits per second (16,200 bps). At the SFOF, the data will be distributed to computers and other specialized processors for reduction and presentation to the experimenters. Simultaneously, spacecraft range and range-rate information will be acquired and relayed to the SFOF for analysis by celestial navigation specialists; spacecraft and experiment status information will be acquired at rates up to 50 bps and distributed to appropriate specialists; and com-



mand changes in the spacecraft sequence will be controlled from the SFOF by mission operations personnel.

In order to provide these capabilities, many new developments will be used to support Mariner Mars 1971. The new command subsystem, which provides for transmission of commands to the spacecraft from any deep space station, under remote control from SFOF locations, will be used for the first time on any planetary project. All deep space stations will be supplied with new telemetry equipment capable of receiving, symbol synchronizing, decoding, and processing high-rate telemetry data. Telecommunications capabilities will allow reception of the full 16.2 kbps at the Goldstone 64-m antenna site, but will limit reception to 2025 bps at the 26-m antenna sites at Goldstone, Australia, Spain, and South Africa. The high-rate data, primarily video information, will be block coded by the spacecraft and transmitted at a maximum rate of 86, 400 binary symbols per second. The new telemetry equipment, a prototype of which was used at Goldstone for the Mariner Mars 1969 encounter, will perform a symbol synchronization and correlation (decoding) function to reproduce the original data stream (16.2 kbps).

A new 50-kbps digital wideband communications link will carry the data from the Goldstone 64-m antenna site to the SFOF. All high-speed data links between the deep space stations and the SFOF will be upgraded from 2400- to 4800-bps capability. This increased communications capability will allow real-time transmission of all data to the SFOF.

Once the data arrives at the SFOF, it will encounter a new data processing complex, consisting of an IBM 360/75 computer coupled with a Univac 1108 computer. The 360/75 system is used to generate, send, and verify spacecraft commands, to receive and process engineering telemetry and tracking data, and to produce system data records. Mission operational control programs, such as AMPS, and science programs, such as LIBSET and IRIS, also run in the 360/75. All navigation programs are operated in the UNIVAC 1108. The UNIVAC MTC system will receive science telemetry data during orbital operations and will provide real-time image data processing, displays of the non-imaging science data and will produce the project telemetry master data records.

This combination of data systems and associated printers, cathode-ray tube display devices, and the video data conversion equipment in MTVS will provide both real-time and non-real-time presentation to the experimenters, spacecraft analysts, and navigation analysts.

The ground data handling for Mariner Mars 1971 is shown in Figure 24; the shaded boxes represent computers.

The following major MOS programs (excluding MTC programs, supporting analysis programs, and the large software operating systems in the various computers) have been developed for the 360/75 and the 1108 systems:

(1) **ICG (Injection Conditions Generator Program).** Computes injection time, velocity, radius; coordinates flight path angle and launch azimuth, using launch time and a polynomial approximation equation.

(2) **DPTRAJ (Double Precision Trajectory Program).** Integrates equations of spacecraft motion from epoch to

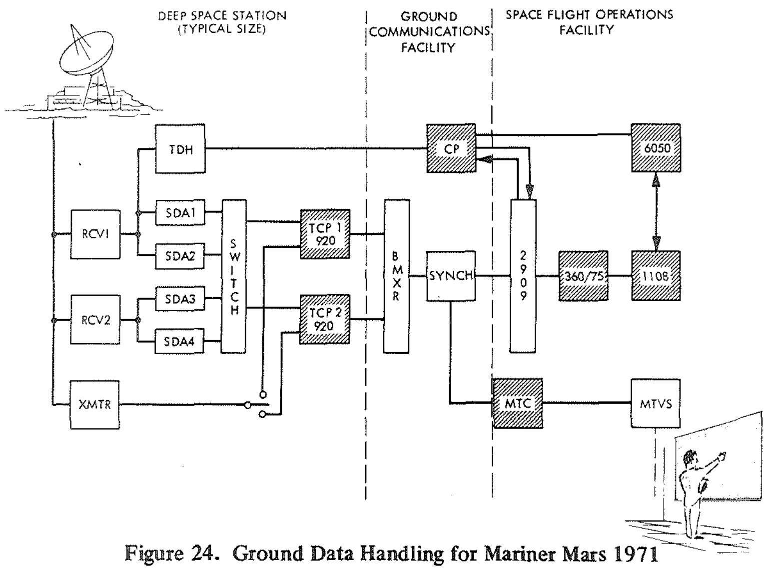


Figure 24. Ground Data Handling for Mariner Mars 1971

desired point using input from ODE, ICG, or nominal data. Provides listings and SAVE tape of position, velocity, look angles, DSS view periods, etc.

(3) **ODE (Orbit Data Editor).** Prepares DP orbit data file from the real time tracking data master file by selection, compression, correction, or calibration, etc.

(4) **DPODP (Double Precision Orbit Determination Program).** Calculates best orbit from ODE file data using weighted least-squares trajectory fit. Maps statistical errors to encounter. Also may solve for physical constants, DSS locations, perturbing forces, etc.

(5) **SATODP (Satellite Orbit Determination Program).** Calculates and maps trajectory and errors as above from orbital data located in the ODE file.

(6) **MOPS (Maneuver Operations Programming System).** Calculates maneuver capabilities, maneuver values and commands required for midcourse maneuver, orbit insertion, and orbit trim.

(7) **POGASIS (Planetary Observation Geometry and Science Instruments Scan Program).** Determines the orbital science strategy that optimizes science data return and computes for the spacecraft the required scan platform angles and instrument viewing times. Conversely, computes actual coverage and observation conditions based on data received from SPOP (platform orientation and time).

(8) **AMPS (Adaptive Mode Planning System).** Automates the operation of a set of programs, i.e., POGASIS, SPOP, SCISIM, COMGEN, SOEGEN, required for the adaptive planning of orbital operations (see Figure 25).

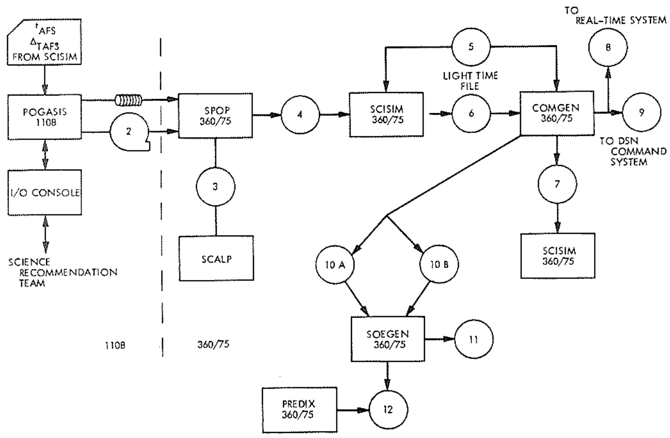


Figure 25. Adaptive Mode Planning System

(9) **LIBSET (Science Library Index System).** Automates the operation of a set of programs, i.e., POGASIS, SPOP, SCISIM, required for producing the science library index.

(10) **COMGEN (Command Generation Program).** Assembles and checks CC&S programs, simulates the CC&S action on the program, and forms spacecraft command messages for loading CC&S memory from this assembly or other sequences provided by SCISIM and SPOP.

(11) **SCISIM (Science Subsystem Event Simulator Program).** Generates flight command subsystem and CC&S commands and timing for COMGEN in order to accomplish specified data automation subsystem sequences. Simulates data automation subsystem sequencing, and predicts time of occurrences of actual science events in any specified time base.

(12) **SOEGEN (Sequence of Events Generator).** Generates and displays a time-ordered sequence of events from file or card input, with capability to display or output by mission, tracking station number, etc.

(13) **SPOP (Scan Platform Operations Program).** Provides commands to COMGEN based on data received from SCALP or POGASIS. Determines best estimate of platform positioning angles from data received from SCALP.

(14) **IRIS (Infrared Interferometer Spectrometer Program).** Accepts experiment data record tape of engineering and interferogram data, processes data and parity information to provide spectral plots and listings, instrument coverage, and performance.

(15) **SCILIB (Science Library Program).** Provides an index of science measurements for all instruments, including coordinates of planetary "footprints", slant ranges, and illumination angles.

(16) **UVS/IRR (Ultraviolet Spectrometer/Infrared Radiometer Program).** Provides calibration and formatting of ultraviolet spectrometer and infrared radiometer science telemetry data for display purposes.

(17) **OCCULTATION (Occultation Science Program).** Accepts received doppler tracking data, calculates residuals from SATODP or predicted values, and analyzes the data in relation to the spacecraft trajectory derived from DPTRAJ to compute planetary atmospheric parameters.

(18) **PSOP (Propulsion Subsystem Operations and Performance Program).** Predicts subsystem performance based on propellant/pressurant subsystem analysis; spacecraft mass distribution and thrust vector orientation based on gimbal actuator pre-aim data.

(19) **TPAP (Telecommunications Performance and Prediction Program).** Computes predicted channel performance from antenna pattern data, spacecraft and ground system characteristics and trajectory data. Computes actual performance from real-time telemetry data and ground system, and compares with predicted data.

(20) **CELREF (Celestial Reference Program).** Computes sensor performance vs clock angle; lists acquirable objects using sensor characteristics and trajectory data. Calculates spacecraft attitude relative to Sun, planets, or selected stars; and outputs cone and clock angles of celestial objects.

(21) **SCALP (Scan Calibration Program).** Provides corrections to SPOP for improving accuracy of scan platform pointing, based on calibration data derived from television pictures of stars and actual scan platform angles.

Because of the complexity of the missions and the requirement for rapid processing of such great amounts of data, greater use will be made of electronic computers for Mariner Mars 1971 than for any previous planetary project. During orbital operations, these electronic computers will perform about 36 billion computations each day 7 days per week. Before electronic computers were developed, the calculations would have been performed on an electromechanical desk computer. In an 8-hour shift, one person operating a desk computer can perform about 2000 equivalent computations. Just to perform the computations, without considering the insurmountable problem of transmitting data between desk computers, would require about 18 million persons operating desk calculators, or about one-quarter of the entire labor force of the United States.

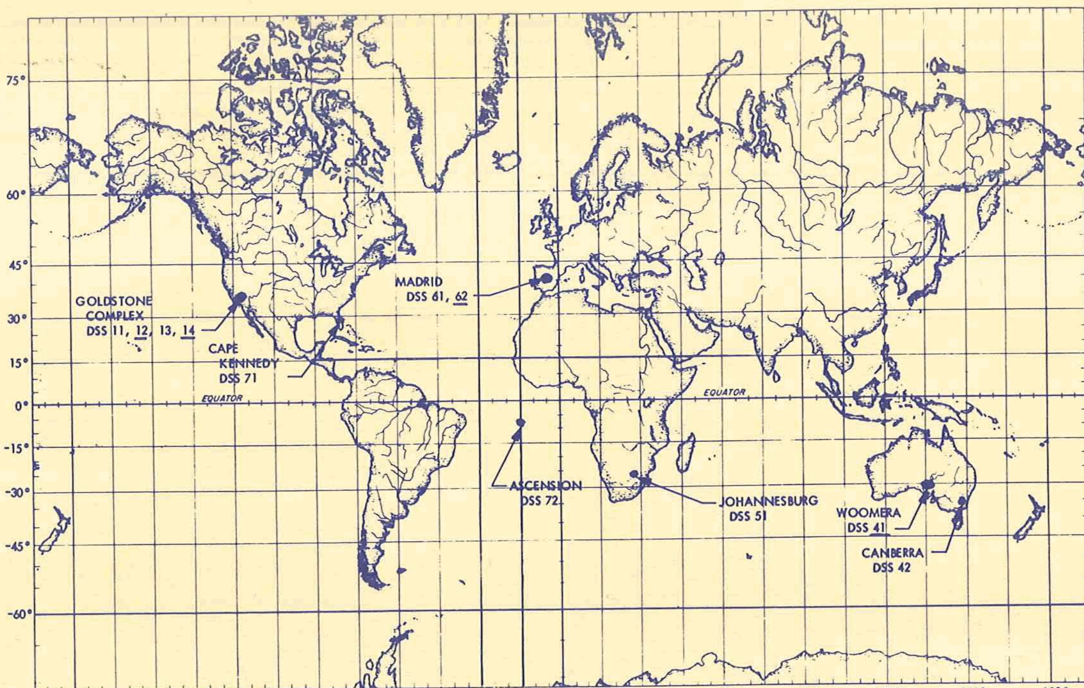
SCIENCE ANALYSIS

Science analysis starts with the first data obtained from the science instruments. The raw (as received from the spacecraft) data will provide the information for determining preliminary results used in operations planning and in initial science reporting. As soon as tape-recorded copies of data from the instruments are available to the investigators, data processing begins. During the processing, communication, spacecraft, and instrument effects are eliminated from the data. Processing varies in complexity for each experiment; it may require a few days or it may take months to complete. The completed file of processed (reduced, or decalibrated) data is the reduced data record, which is used for science analysis and the generation of the final science experiment results.

Results from both raw and reduced data are used to formulate recommendations for mission operations. Information derived from each experiment is used to make a daily recommendation to the Chief of Mission Operations regarding whether to proceed or deviate from operations already planned. The decisions for the recommendations are made by members of the Science Recommendation Team, with the goal of maximizing the overall science results to be obtained from the mission.

Because the scientific objectives can best be met by adaptive mission operations, preparations have been made to process the science data from the beginning of operations. The processed data will be used as soon as available for analysis. The total complement of 56 investigators will be involved in the analysis effort; they will be supported by an almost equal number of scientific and technical personnel.

Science results derived from the analyses will be presented initially to the general public through information releases and press conferences. Project science reports that contain observations from all experiments will be published for the technical community. These reports will be published 1 month, 9 months, and 18 months after the start of the orbital missions. Results from each experiment will be documented by the investigators and published in scientific journals, for the scientific community, as soon as completed. It is expected that reporting based on the data obtained from the missions will continue for several years.


GOLDSTONE COMPLEX
DSS 11, 12, 13, 14
CAPE KENNEDY
DSS 71
MADRID
DSS 61, 62
ASCENSION
DSS 72
JOHANNESBURG
DSS 51
WOOMERA
DSS 41
CANBERRA
DSS 42
EQUATOR
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